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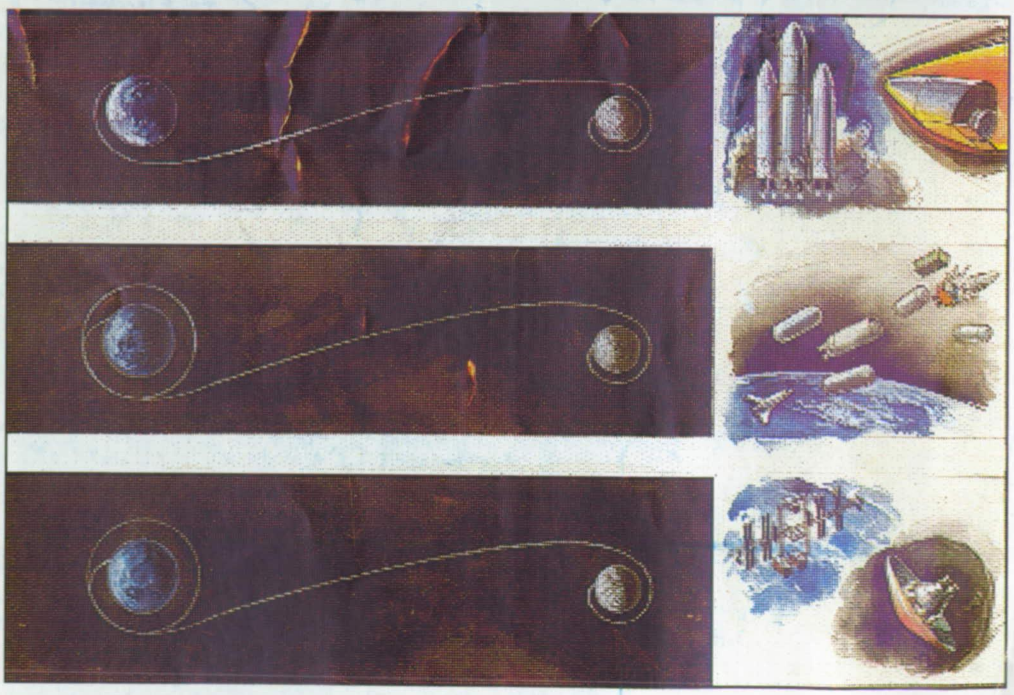
Space Transfer Vehicle
Concepts and Requirements Study



Volume II Final Report
Book 2 System & Program Requirement Trade Studies

April 1, 1991
NAS8-37855

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PREPARED BY:	Gary Weber G.A. Weber	2-3932	3/22/91
SUPERVISED BY:	TJ Vinopal T.J. Vinopal	2-3900	3/26/91
APPROVED BY:	SD Goo S.D. Goo	2-3900	3/26/91

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SPACE TRANSFER VEHICLE CONCEPTS AND REQUIREMENTS STUDY

Phase I Final Report
Volume II, Book 2
System & Program Requirements Trade Studies
D180-32040-2
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By
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Seattle, Washington 98124

FOREWORD

This final report of the first phase of the Space Transfer Vehicle (STV) Concept and Requirements Study was prepared by Boeing for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center in accordance with Contract NAS8-37855. The study was conducted under the direction of the NASA Contracting Officer Technical Representative (COTR), Mr Donald Saxton from August 1989 to November 1990, and Ms Cynthia Frost from December 1990 to April 1991.

This final report is organized into the following seven documents:

Volume I EXECUTIVE SUMMARY

Volume II FINAL REPORT
Book 1 - STV Concept Definition and Evaluation
Book 2 - System & Program Requirements Trade Studies
Book 3 - STV System Interfaces
Book 4 - Integrated Advanced Technology Development

Volume III PROGRAM COSTS ESTIMATES
Book 1 - Program Cost Estimates (DR-6)
Book 2 - WBS and Dictionary (DR-5)

The following appendices were delivered to the MSFC COTR and contain the raw data and notes generated over the course of the study:

Appendix A	90 day "Skunkworks" Study Support
Appendix B	Architecture Study Mission Scenarios
Appendix C	Interface Operations Flows
Appendix D	Phase C/D & Aerobrake Tech. Schedule Networks

The following personnel were key contributors during the conduct of the study in the disciplines shown:

Study Manager	Tim Vinopal
Mission & System Analysis	Bill Richards, Gary Weber, Greg Paddock, Peter Maricich
Operations	Bruce Bouton, Jim Hagen

Key Contributors Continued

Configurations	Richard Kolesar, Craig Hosking, George Dishman, Mike Furlong, Bob Kiliz, Jack Olson
Propulsion	Wayne Patterson, Noel Christensen, Phillip Knowles
Avionics	Rich Flannagan, Tim Mosher, Carl Malec
Structures	Peter Rimbos, Martin Gibbins
Electrical Power	Chris Johnson
Cryo Fluid Management	Ogden Jones, Jere Meserole
Mass Properties	Jeff Cannon, David Raese, Karl Heilborn
Aerothermodynamics	Richard Savage, Peter Keller
Thermal Protection	Anna Baker, Paul Nedervelt
Guidance	Lisa Skalecki, Jere Bradt
Controls	Mark Castellicio
Performance/Astrodynamics	Ted Hanson, Ralph Risdall, Steve Paris, Mark Martin
Aerodynamics	Stan Ferguson
Crew Systems	Tom Slavin, Brand Griffin, Bill Pogue, Gerry Carr
Station Accomodations	John Palmer, Ron Rao, Carl Case
Cost Estimating	Thom Walters, Al Peffley, Hal Boggs, Jim Owens
Programmatics	Al Peffley, Don Benson, Lori Todd, Bob Croken
Documentation Support	Symantha Rodenbach, Darlene Glubrecht

For further information contact:

Cynthia Frost
NASA MSFC/PT41
MSFC, AL 35812
(205)544-0628

Tim Vinopal
Boeing Aerospace
M/S 8K-52, P. O. Box 3999
Seattle, WA 98124-2499
(206)773-6363

CONTENTS

ACRONYMS

AC	attitude control
ACS	attitude control system
ALS	Advanced Launch System
APU	auxiliary power unit
ASIC	application-specific integrated circuit
ATC	active thermal control
ATDRSS	advanced TDRSS
BIT	built-in test
BOLT	Boeing Lunar Trajectory Program
CASE	computer-aided software engineering
CNDB	civil needs database
CNSR	comet nucleus sample return
CT	communications and tracking
CTE	coefficient of thermal expansion
DAK	double aluminized Kapton
DDT&E	design, development, test, and evaluation
(delta) T	change in event duration
(delta) V	change in velocity
DoD	Department of Defense
DMR	design reference missions
DRS	design reference scenario
DSN	deep space network
ECLSS	environmental control and life support system
EOS	Earth observing system
EPS	electrical power system
ESA	European Space Agency
ETO	Earth to orbit
EVA	extravehicular activity
FC	fluid control
FEPC	flight equipment processing center
FOG	fiber-optic gyro
FSD	full-scale development
GB	ground based
GC	guidance control
GEO	geosynchronous orbit
GLOW	gross liftoff weight
GNC	guidance, navigation, and control
GO	ground based, on orbit
GPS	global positioning system
GSE	ground support equipment
HEI	Human Exploration Initiative
HEO	high Earth orbit
HESR	Human Exploration Study Requirements
HLLV	heavy lift launch vehicle

ICI	Integrated Systems Incorporated
ILD	injection laser diode
IMU	inertial measurement unit
IUS	Inertial Upper Stage
IVA	intravehicular activity
JPL	Jet Propulsion Laboratory
JSC	Johnson Space Center
KSC	Kennedy Space Center
LAD	liquid acquisition device
LAN	local area network
LCC	life cycle cost
LCD	liquid crystal display
L/D	lift to drag
LECM	lunar excursion crew module
LED	light-emitting diode
LEO	low Earth orbit
LES	launch escape system
LEV	lunar excursion vehicle
LLO	low lunar orbit
LMS	lunar mission survey
LO	lunar orbiter
LOD	lunar orbit direct
LOI	lunar orbit injection
LOR	lunar orbit rendezvous
LOX/LH	liquid oxygen/liquid hydrogen
LTS	lunar transportation system
LTV	lunar transfer vehicle
MEOP	maximum expected operating pressure
MET	mission elapsed time
MEV	Mars excursion vehicle
MLI	multilayer insulation
MPS	main propulsion system
MSFC	Marshall Space Flight Center
MTPE	mission to planet Earth
MTV	Mars transfer vehicle
NEP	nuclear energy propulsion
NPSH	net positive suction head
NTR	nuclear thermal rocket
ORU	orbit replaceable unit
P/A	propulsion/avionics
PC	propulsion control
PCM	parametric cost model
PDT	product development team
PODS	passive orbital disconnect strut
PSS	planet surface system
PVT	pressure-volume-temperature

RCS	reaction control subsystem
RFP	request for proposal
RLG	ring laser gyros
RMS	remote manipulator system
RTV	room temperature vulcanizating
SB	space based
SEI	Space Exploration Initiative
SEP	solar energy propulsion
SEU	single-event upset
SG	space/ground
SIP	strain isolation pad
SIRF	spaceborne imaging radar facility
SIRTF	Space IR Telescope Facility
SLAR	side-looking aperture radar
SOS	silicon on sapphire
SRM	solid rocket motor
SSF	Space Station Freedom
STIS	Space Transportation Infrastructure Study
STS	space transportation system
STV	Space Transfer Vehicle
TDRSS	tracking and data relay satellite system
TEI	trans-Earth injection
TLI	translunar injection
TMI	trans-Mars injection
TPS	thermal protection system
TVC	thrust vector control
TVS	thermodynamic vent system
USRS	Upper Stage Responsiveness Study
VHM	vehicle health monitoring
VHMS	vehicle health management system
ZLG	zero lock gyro

2-1.0 SYSTEM AND PROGRAM TRADE STUDIES

Introduction. During the 90-day study, support was provided to NASA in defining a point-of-departure STV. The resulting STV concept was performance optimized with a two-stage LTV/LEV configuration. Appendix A reports on the effort during this period of the study. From the end of the 90-day study until the March Interim Review, effort was placed on optimizing the two-stage vehicle approach identified in the 90-day effort. After the March Interim Review (IR#2), the effort was expanded to perform a full architectural trade study with the intent of developing a decision database to support STV system decisions in response to changing SEI infrastructure concepts. Several of the architecture trade studies were combined in a System Architecture Trade Study. In addition to this trade, system optimization/definition trades and analyses were completed and some special topics were addressed. Program- and system-level trade study and analysis methodologies and results are presented in this section. Trades and analyses covered in this section are:

1. System Architecture Trade Study.
   a. Number of stages.
   b. Crew module approaches.
   c. Basing approaches.
   d. lunar approach trajectory.
   e. Aerobrake versus all-propulsive return.
   f. Use of droptanks versus propellant tankers.
2. Evolution.
3. Safety and abort considerations.
4. STV as a launch vehicle upper stage.
5. Optimum crew and cargo split.

The subsystem trade studies are presented in volume II, book 1, section 3.0 (i.e., volume II, 1-3).

2-1.1 SYSTEM ARCHITECTURE TRADE STUDY

This section covers the System Architecture Trade Study. The overall trade study is introduced and each of the six individual architecture trade studies that

were combined in the system architecture trade are discussed in terms of trade issues and options. Next, the process used to determine the combinations of trade options to be examined and the process used to characterize these combinations is presented. Finally, the methods used to evaluate the options, the trade results, and some selected sensitivity data are discussed.

The System Architecture Trade Study was a major effort of the STV study and combined several architecture trades into an overall architecture trade study. Several of the architecture trades were interdependent, so it was felt that a combined trade could account for the interactions by providing evaluations of one trade across different options of other interdependent trades. In this method, the best combination of architectural options could be determined. Evaluation criteria and criteria weighting against which the options were evaluated consisted of cost, 50% weighting; margins and risk, 30% weighting; other mission capture, 15% weighting; and benefits to Mars, 5% weighting.

The options defined for the six architecture trades were combined in a matrix resulting in over 400 possible architectures. Groundrules and assumptions were applied to reduce these combinations to 94 architectures for which performance and mission scenarios were developed. Based on this work, 29 scenarios were selected and initially assessed against the cost and margins and risk evaluation criteria to determine trending. Based on the observed trends, 13 additional scenarios were initially included with one being added later. The resulting 43 scenarios were fully evaluated against the four evaluation criteria to determine the preferred architectures. Figure 2-1.1-1 provides an overview of the System Architecture Trade Study process.

2-1.1.1 Architecture Trades

The architecture trades that were combined in the System Architecture Trade Study consisted of number of stages, crew module approach, basing location, lunar approach trajectory, aerobraked versus all-propulsive return, and use of droptanks versus propellant tankers. The latter two trades were relevant to the space-based cases only. Inherent in these trades were examinations of reusability. For example, in the space-based cases where propellant tankers were used, the entire vehicle was reused, and in the cases where ground

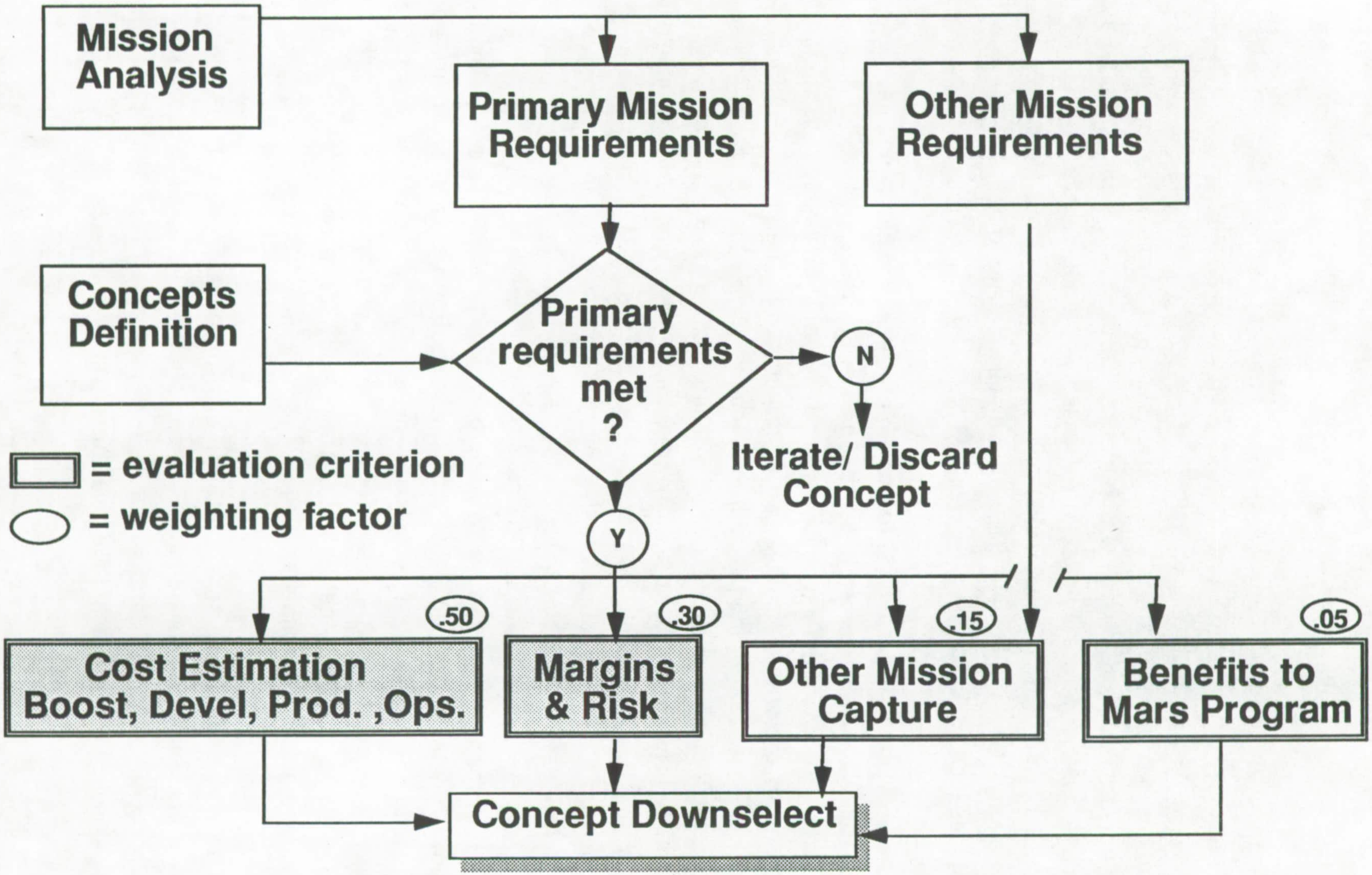


Figure 2-1.1-1. Approach to STV Concept Selection

basing was used, only the crew module was reused. Figure 2-1.1.1-1 provides an overview of the architecture trades and trade options discussed below.

Number of Stages. Stages were defined as propulsion plus tankage. Use of droptanks or sets of droptanks was designated as a 0.5 stage. Thus, a 2.5 stage had two sets of engines with tankage and used a set or sets of droptanks. For the cases in which a low lunar orbit (LLO) node was used for storage of elements while the mission was performed in 0.5 stage increments, 1.5 to 4 stage options were examined. In the cases where the vehicle went directly to the lunar surface, 1.5 to 3 stage options were examined. Figures 2-1.1.1-2 through 2-1.1.1-6 show the staging assumptions used to define the use of the stages for each option.

Crew Module Approach. Three options for crew module configurations and operations were evaluated and are shown in Figure 2-1.1.1-7. The dual crew module approach was the option selected for the 90-day study baseline. In this case, a transfer crew module carried the crew from Earth to LLO. This crew module was a larger module with the required volume for the trip duration and carried radiation shielding (water). Based in LLO was the excursion crew module. This would be a smaller crew module, with no radiation shielding, which would mate with the transfer crew module in LLO for crew transfer, transport the crew from LLO to the lunar surface, and after the mission, return the crew to the transfer crew module for return to Earth.

The hybrid crew module was an approach similar to the dual crew module; however, the excursion cab would return to Earth/LEO between missions instead of being LLO based between missions as in the dual approach. Note that both the dual and hybrid crew modules depend on a lunar trajectory and operations approach that uses LLO as a node for mass storage during the mission on the lunar surface.

Use of a single crew module was the third option examined. This option has one crew module that performs the entire mission. The single crew module approach could be used either in conjunction with an LLO node or in conjunction with a direct to the lunar surface lunar approach.

Goal: To downselect to a few winning concepts

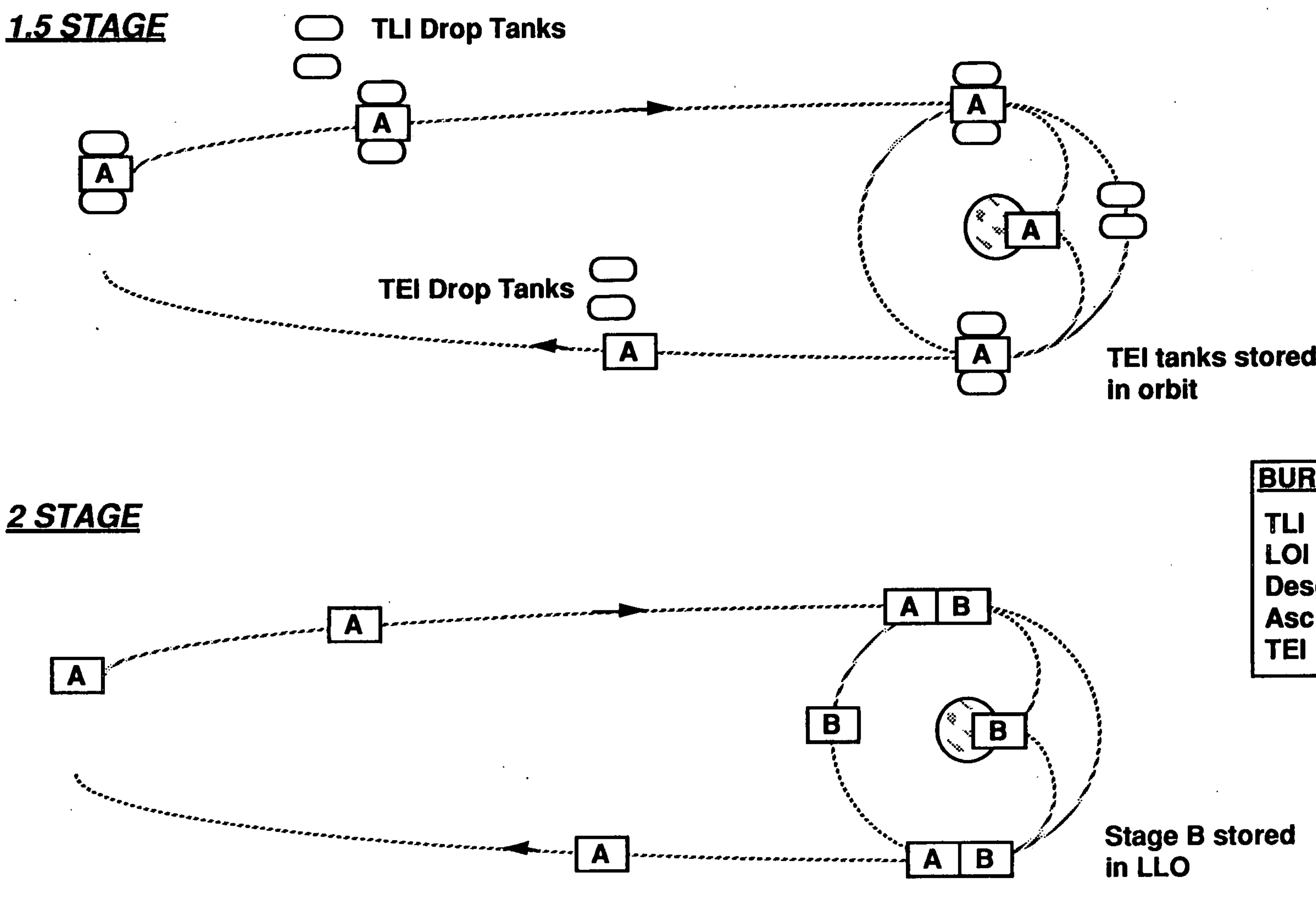
Starting Point: 6 Concurrent Architecture studies;

Trade	Options
ALL CASES	
Basing location	Ground Based single & multiple launch, Space Based @ SSF, and Mixed space / ground
Transfer Orbit	Lunar Orbit Rendezvous, Direct to Lunar Surface (Lunar Orbit Direct since identified)
Number of Stages	1.5 (.5 stage = drop tanks), 2, 2.5, 3, 3.5, 4 stages
Crew Modules	Single, Dual (transfer & excursion), or Hybrid
SPACE BASED ONLY	
LEO Circularization	All Propulsive or Aerobrake
Propellant Delivery	Tankers, Wet Drop Tanks, or Wet Stages

These provide over 400 distinct combinations, reduced to ~94 for analyses, with full evaluation completed on key 43.

Figure 2-1.1.1-1. System Architecture Trade Studies - Overview

Staging approaches for options going through LLO, regardless of Earth/LEO basing/return



BURN	ΔV-m/s
TLI	3300
LOI	1100
Desc	2000
Asc	1900
TEI	1100

Figure 2-1.1.1-2. 1.5- and 2-Stage Overview - LLO Node

Staging approaches for options going through LLO, regardless of Earth/LEO basing/return

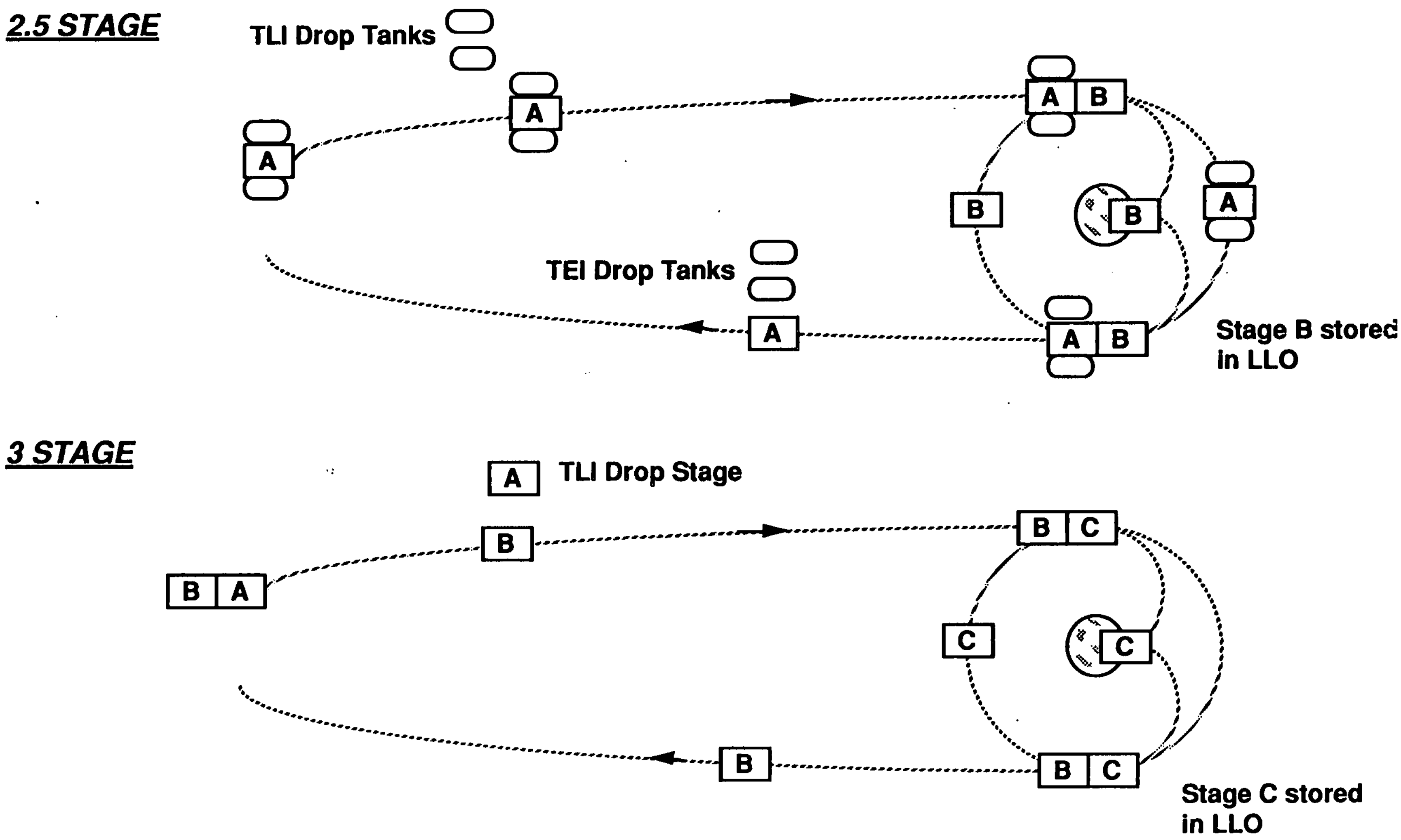
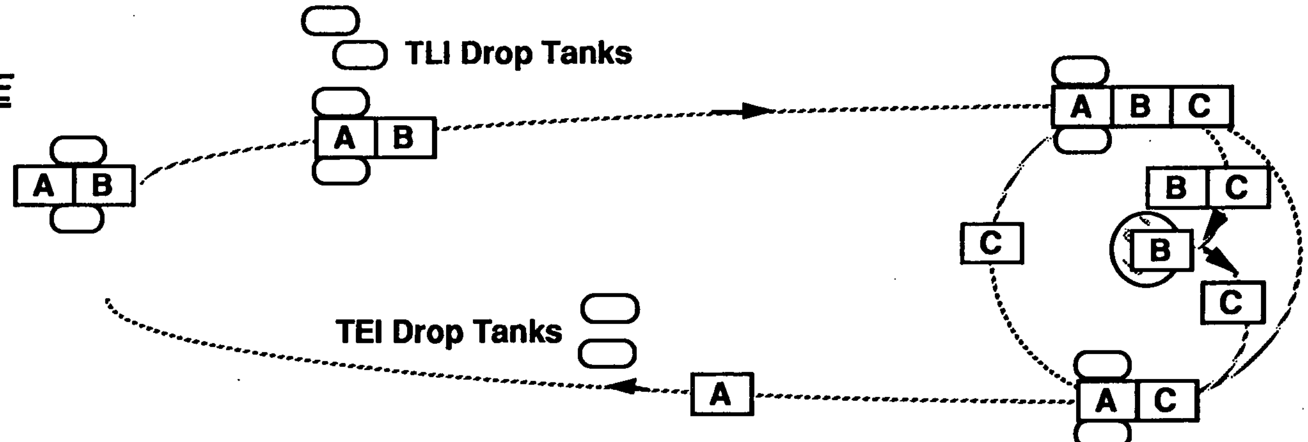


Figure 2-1.1.1-3. 2-5- and 3-Stage Overview - LLO Node

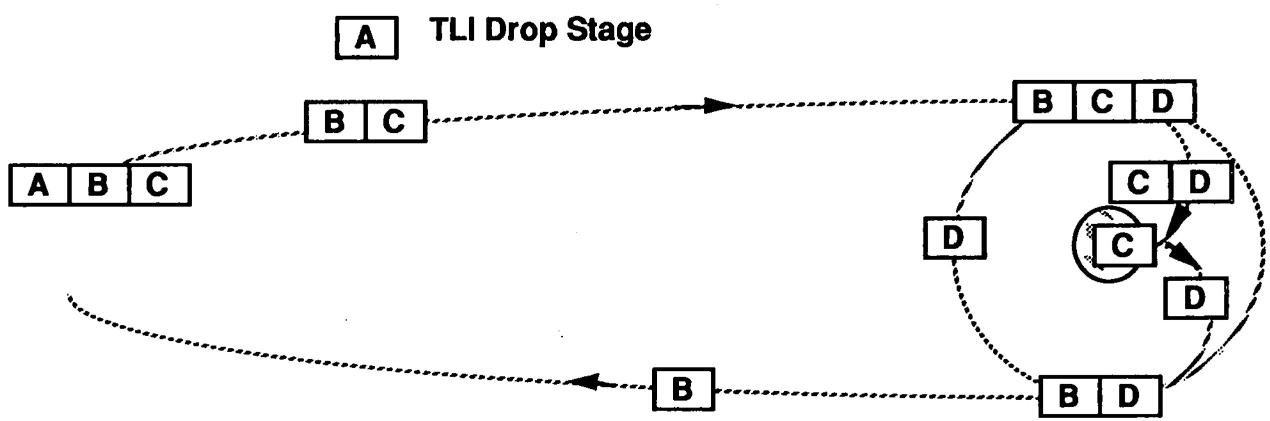
Staging approaches for options going through LLO, regardless of Earth/LEO basing/return

3.5 STAGE



Stage C stored in LLO, stage B is a descent stage from Earth, expended on LS each time

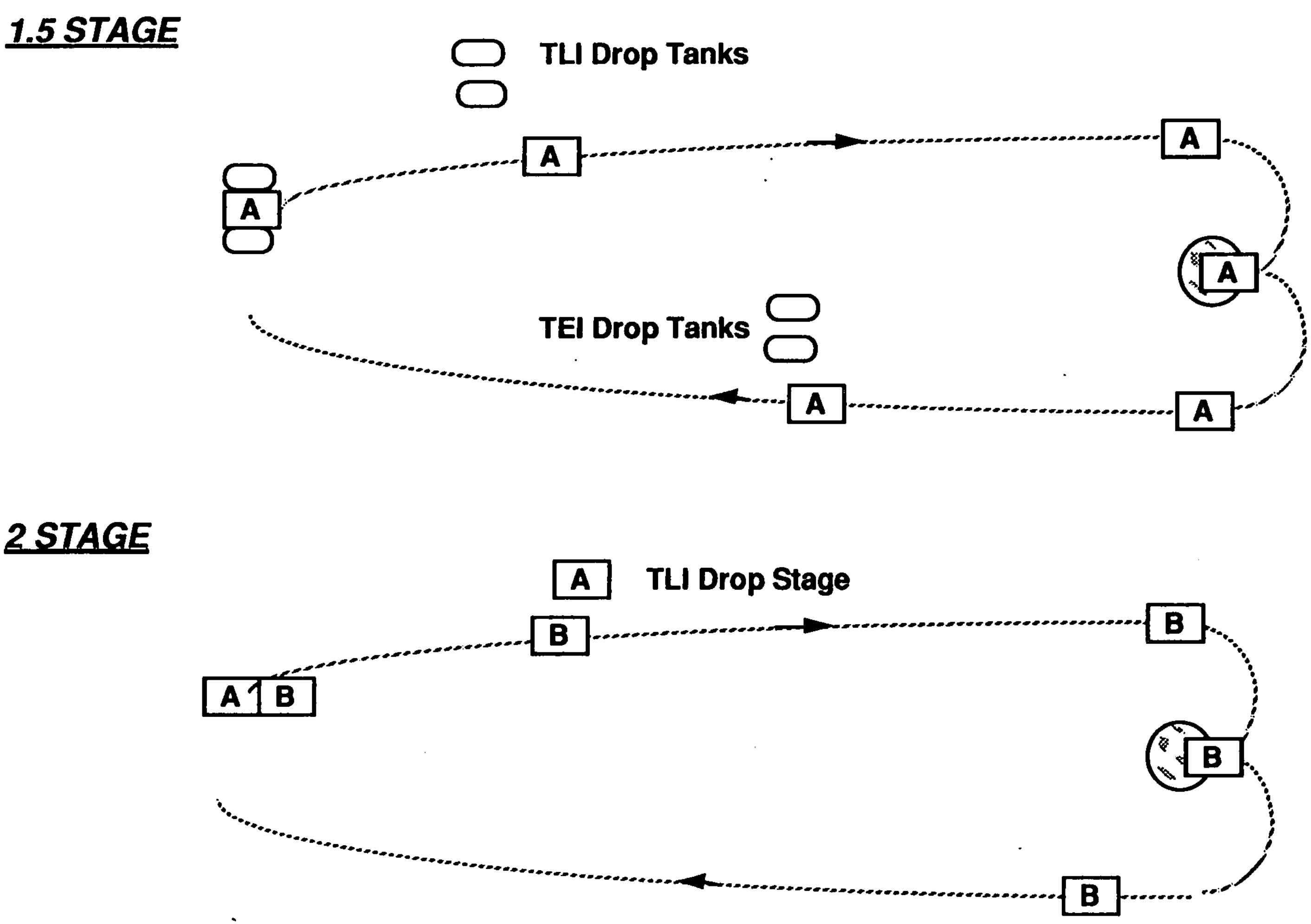
4 STAGE



Stage D stored in LLO, stage C is a descent stage from Earth, expended on LS each time

Figure 2-1.1.1-4. 3.5- and 4-Stage Overview - LLO Node

Staging approaches for options going direct to the LS, regardless of Earth/LEO basing/return



BURN	ΔV-m/s
TLI	3200
LS Landing	2510
LS Takeoff	2510

Figure 2-1.1.1-5. 1.5- and 2-Stage Overview - Direct to LS

Staging approaches for options going direct to the LS, regardless of Earth/LEO basing/return

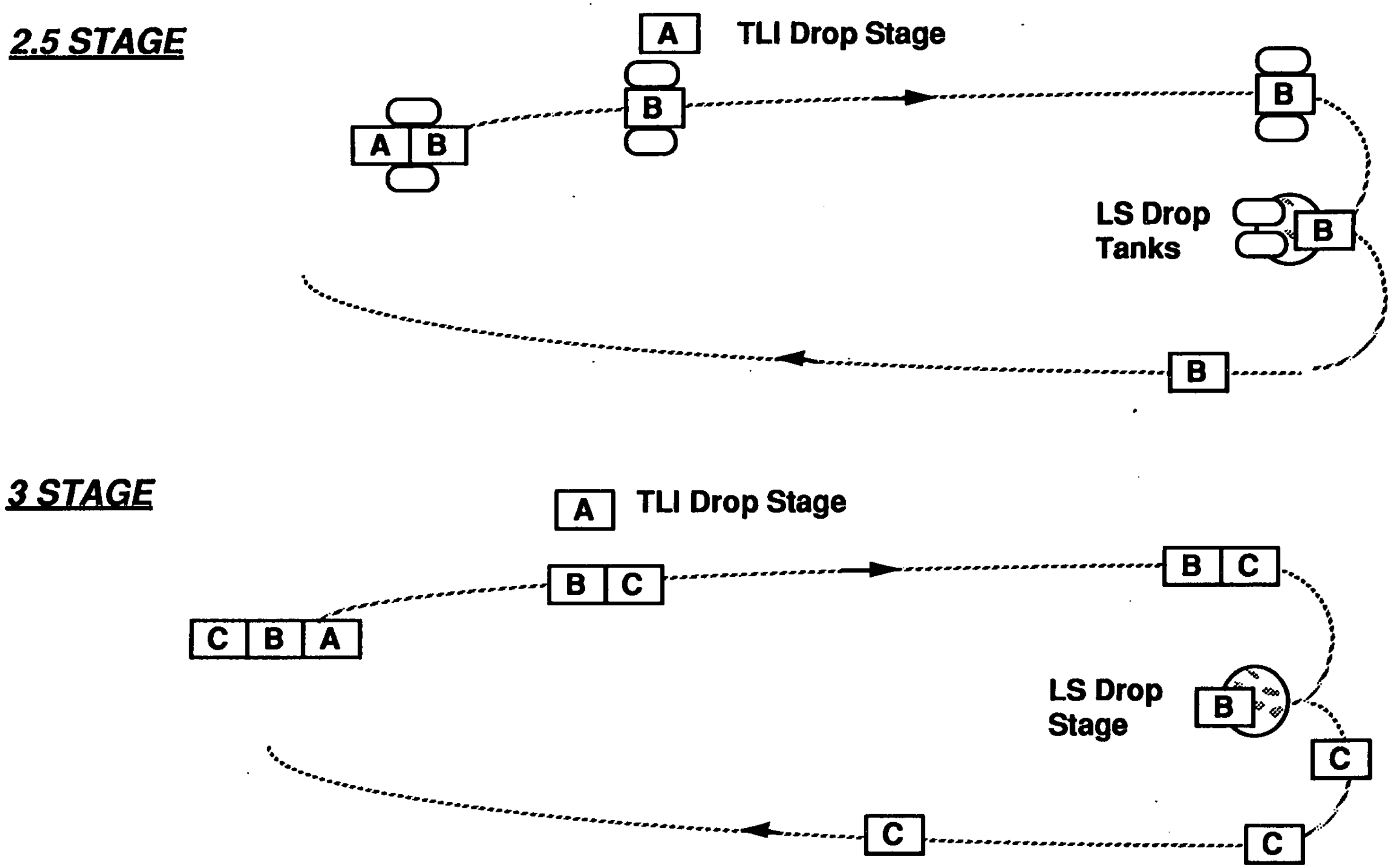
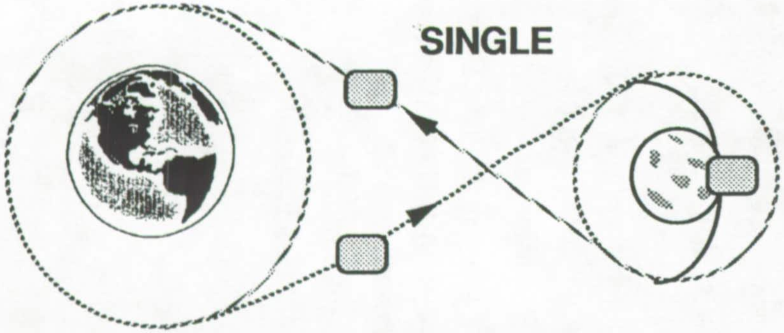
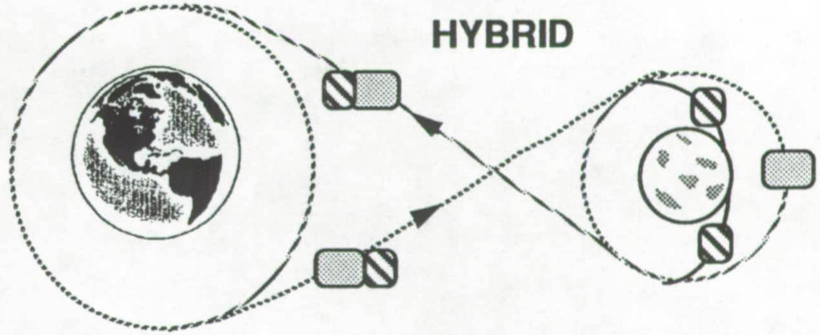


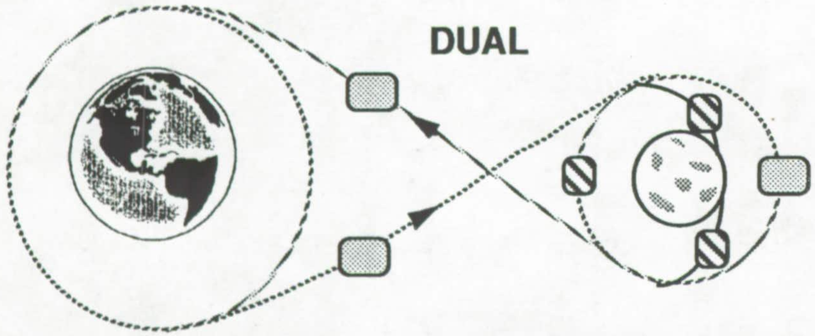
Figure 2-1.1.1-6. 2.5- and 3-Stage Overview - Direct to LS



- Single Crew Module performs all phases of mission



- Both crew modules travel from Earth/LEO to LLO and back
- Excursion cab performs all flight control functions including Lunar ascent & descent while transfer cab stays in LLO



- Excursion cab stored in LLO between missions
- Transfer module transfers crew to/from Earth/LEO to LLO
- Crew transfered to excursion module for Lunar descent & ascent and back to transfer module for Earth/LEO return
- Both modules require flight control functions

Single or Transfer module

Excursion module

Figure 2-1.1.1-7. Crew Module Options

Basing Location. Four options for basing, shown in Figure 2-1.1.1-8, were examined. Note that basing was defined as being in the vicinity of Earth/LEO. The options examined with respect to lunar vicinity operations were defined independently in the lunar approach trajectory section of the trade.

Ground-Based (GB), Single Launch. This basing option had the STV entirely ground based. The STV with cargo, crew module, and crew is launched in one launch. This option would require development of a large booster (260 metric ton class). Upon return, the crew module would be ballistically returned to Earth for refurbishment and would be the only element to be reused.

Ground-Based, On-Orbit (GO) Assembly. A variation of the ground-based approach was ground basing with multiple launches. This option has elements launched separately and, through a series of rendezvous and docking maneuvers, the STV and cargo is assembled autonomously in LEO. Again, the crew module would be ballistically returned for refurbishment and reuse with the other elements being expended.

Space-Based (SB). Space basing was the reference for the 90-day study. In this option, the vehicle would depart from and return to a LEO node (SSF-assumed). Cargo, crew, and propellant or propellant tanks would be launched from Earth for each mission. At the SSF, the STV would be refurbished and mated with the Earth-launched elements in preparation for the mission.

Combination Space/Ground Based (SG). A combination basing option was identified. In this scenario, the crew module is a ballistically returned, ground-based module, and space basing is used for the stages. The crew module, crew, cargo, and propellant or propellant tanks would be launched from the Earth for each mission. At the SSF, the STV stage would be refurbished and assembled with the Earth-launched elements. One of the reasons that this approach was defined was that the crew could be directly returned to Earth and the labor-intensive refurbishment of the crew module could take place on the ground.

Lunar Approach Trajectory. The lunar approach trajectory portion of the trade concentrated on the approach and operations in the vicinity of the moon.

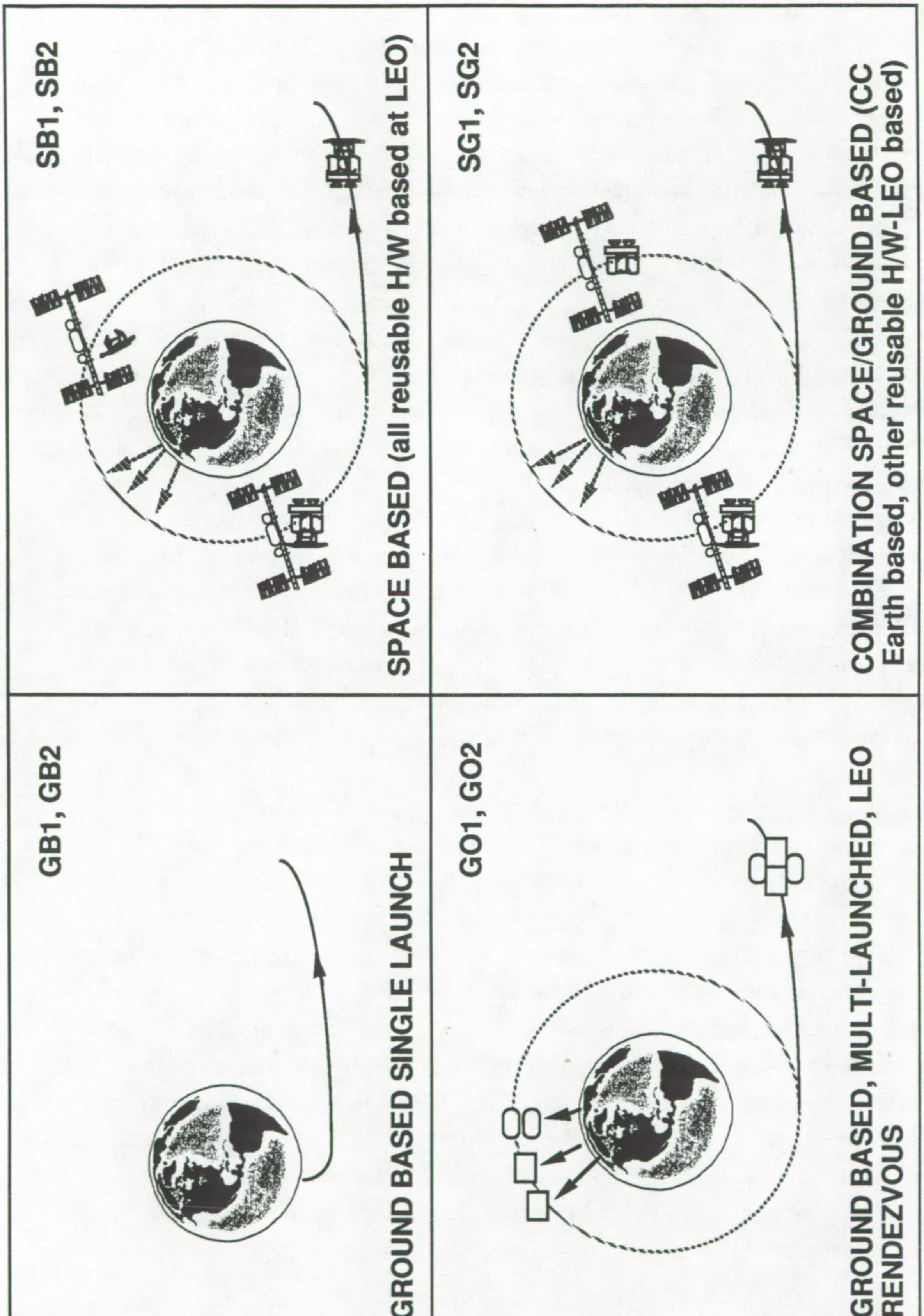


Figure 2-1.1.1-8. Basing Options

Two options (lunar orbit rendezvous (LOR) and lunar surface direct) were evaluated through the System Architecture Trade Study. A third option (lunar orbit direct (LOD)) was identified near the end of the evaluation process. The third identified option was a variation of the direct approach. The direct approach was preferred over LOR based on the evaluation criteria. LOD was similar in terms of evaluation criteria to the direct approach but had better safety and lunar site coverage characteristics and had the lowest ΔV requirements. This option was selected based on the considerations discussed in the following paragraphs.

Lunar Orbit Rendezvous. The LOR approach, shown in Figure 2-1.1.1-9, was selected for the 90-day reference and was the approach used for the Apollo missions. In this option, a LLO node was used for mass storage during the lunar surface missions. Depending on the scenario, Earth-to-LLO transfer and/or return elements were left in a LLO parking orbit while the lunar surface tasks were performed. Upon completion of the lunar surface stay time, the lunar excursion portion of the STV would rendezvous and dock with the elements stored in LLO and the return to Earth would be initiated. Between missions, some scenarios left elements in LLO (e.g., LLO-based excursion stages and/or excursion crew modules) and some scenarios did not depending on the number of stages and crew module approach.

Lunar Surface Direct. The direct approach was a single burn approach where the landing site is targeted and the STV performs a single landing burn. Figure 2-1.1.1-9 provides an overview of this approach.

Lunar Orbit Direct. This approach was conceived during evaluations of the lunar surface direct option to mitigate some of the safety concerns related to the lunar surface direct approach. In this scenario, the STV inserts into an elliptical lunar orbit and then, without leaving anything in orbit, performs a landing burn. The approach assumed, shown in Figure 2-1.1.1-9, would be to burn into the transfer orbit, stay in this orbit for only a portion of a revolution, and then accomplish the lunar landing. The use of a fractional orbit may be ambitious in terms of navigational capability, so the option exists to stay in this elliptical orbit for some number of revolutions prior to landing. This would initially provide time

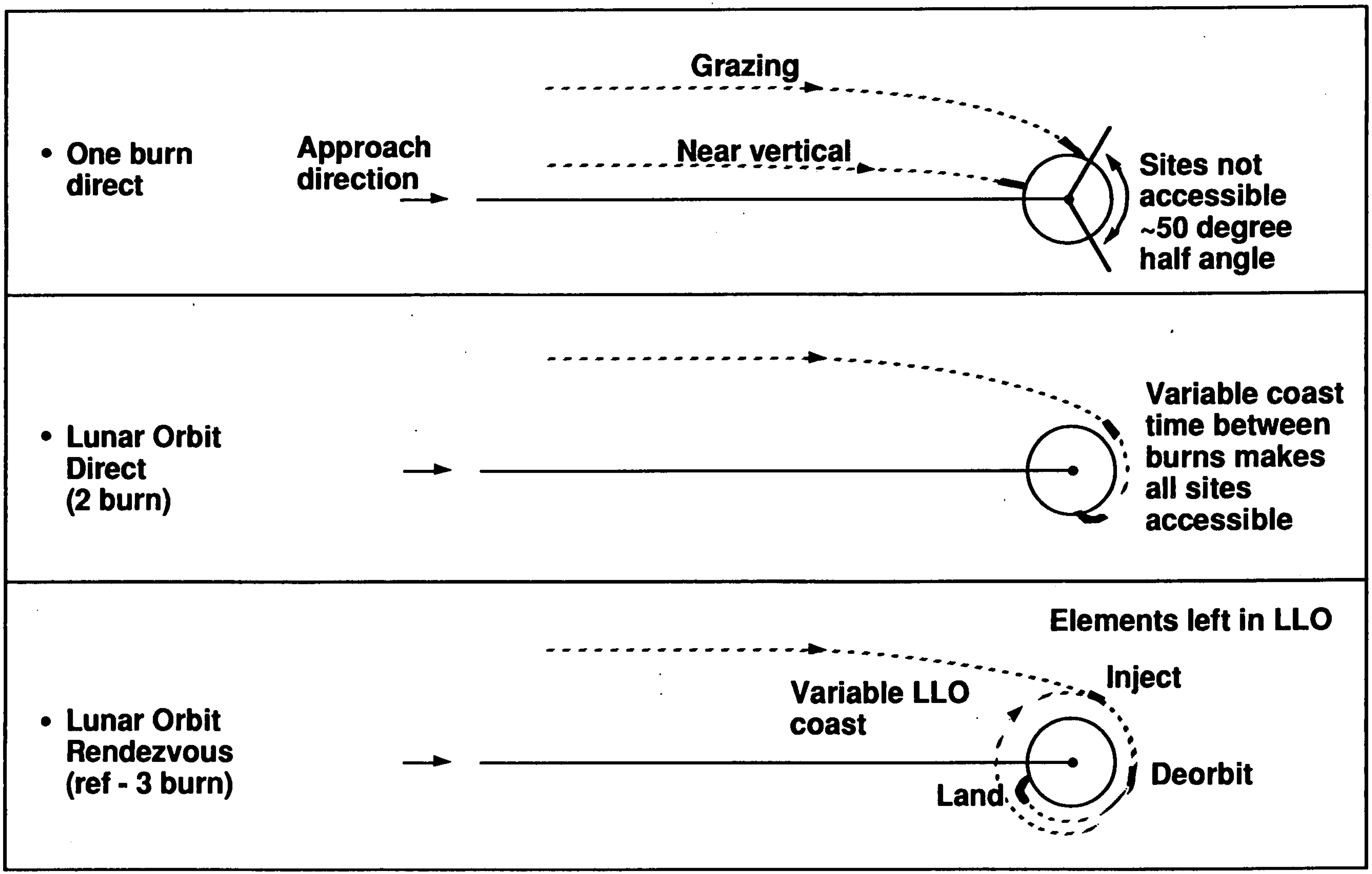


Figure 2-1.1.1-9. Lunar Approach Trajectory Options

for navigation updates while providing a growth path to the fractional orbit approach as navigation capabilities are verified.

Aerobraked Versus All-Propulsive Return. Two approaches were evaluated for return to a space-based transportation node. The all-propulsive return requires a 3,300 m/s ΔV main propulsion system burn for direct insertion into the required LEO. The second option uses an aerobrake to slow down by braking through the Earth's atmosphere. After the STV has been braked to a point of Earth orbit capture, a propulsive maneuver of approximately 310 m/s ΔV is required to circularize in the LEO transportation node orbit.

Droptanks Versus Propellant Tankers. For the space-based cases and lunar missions only, use of propellant tankers versus droptanks was examined. In the case where propellant tankers were available, the entire LEO-based portion of the STV was returned to the SSF for reuse, and propellant from an expendable, Earth-launched propellant tanker was used to refuel for the next mission. The advantage of this approach is that the entire STV could be reused. An additional advantage is that the delivery of propellant to LEO is decoupled from the actual missions. With this approach, ETO propellant delivery launches can be fully manifested and the surplus propellant not required for the mission can be stored for future use. The disadvantages were found in performance. Additional propellant was required to carry the tanks through the entire mission.

With options using droptanks, tanks were expended after the major burns as shown in the stage approach options, Figures 2-1.1.1-2 through 2-1.1.1-6. Replacement droptanks were then launched wet from Earth for assembly with the core vehicle at the SSF. The advantage of this approach is in improved performance similar to the performance gain achieved by staging where unneeded mass is jettisoned as soon as possible. Of course, the droptanks have to be replaced for each mission.

2-1.1.2 System Architecture Trade Study Methodology

This section covers the process used in the architecture trade: the selection of combinations to be examined; the development of mission scenarios; and the

evaluation of the scenarios against the criteria. An overview of this process is shown in Figure 2-1.1.2-1.

2-1.1.2.1 Scenario Selection and Development

The definition of combinations to be examined started with an assessment of orbital options based on use of different basing locations and transportation nodes. Figure 2-1.1.2.1-1 shows the different paths that the STV could follow through the mission. Options that had, for example, the STV leave from Earth and return to the SSF were defined but not considered. These types of missions could be used for replacement of expended assets for a space-based case; however, only steady-state orbital options were considered. Another assumption was that use of an LLO node on the way to the lunar surface would be accompanied by use of the LLO node on the return trip. This was assumed because use of an LLO node implied mass storage, which would have to be picked up prior to return.

The orbital options were then combined in a matrix with the various architecture options to identify possible scenarios. See Figure 2-1.1.2.1-2 for a definition of the terminology used for the scenarios. Evaluation and deletion of possible approaches took place at several levels. While filling out the initial scenario matrix (Figure 2-1.1.2.1-3), the groundrules established were:

1. Crew/cargo vehicles with more than three stages and expendable cargo vehicles with more than two stages were not considered for direct to the lunar surface trajectories.
2. For direct to the lunar surface trajectories, only a single crew module was considered.

After the initial architecture matrix was populated, a second set of groundrules and assumptions was used to further reduce possible scenarios. The scenarios deleted through application of this set of groundrules are shown in Figure 2-1.1.2.1-3 in reduced size, plain text. These were reduced based on the following:

1. Expendable cargo missions do not stop in LLO (delete EC3 and EC4).

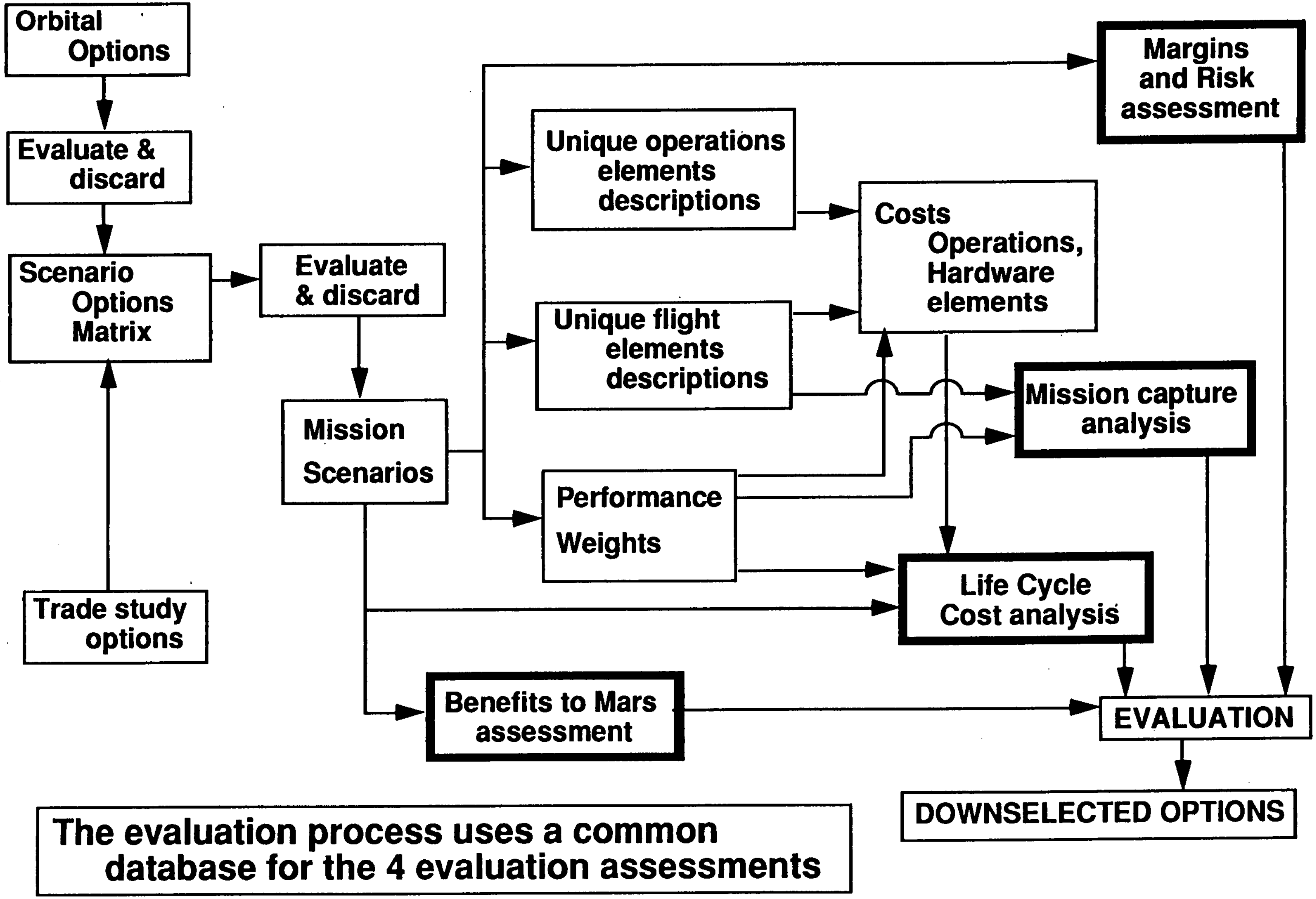


Figure 2-1.1.2-1. System Architecture Evaluation Flow

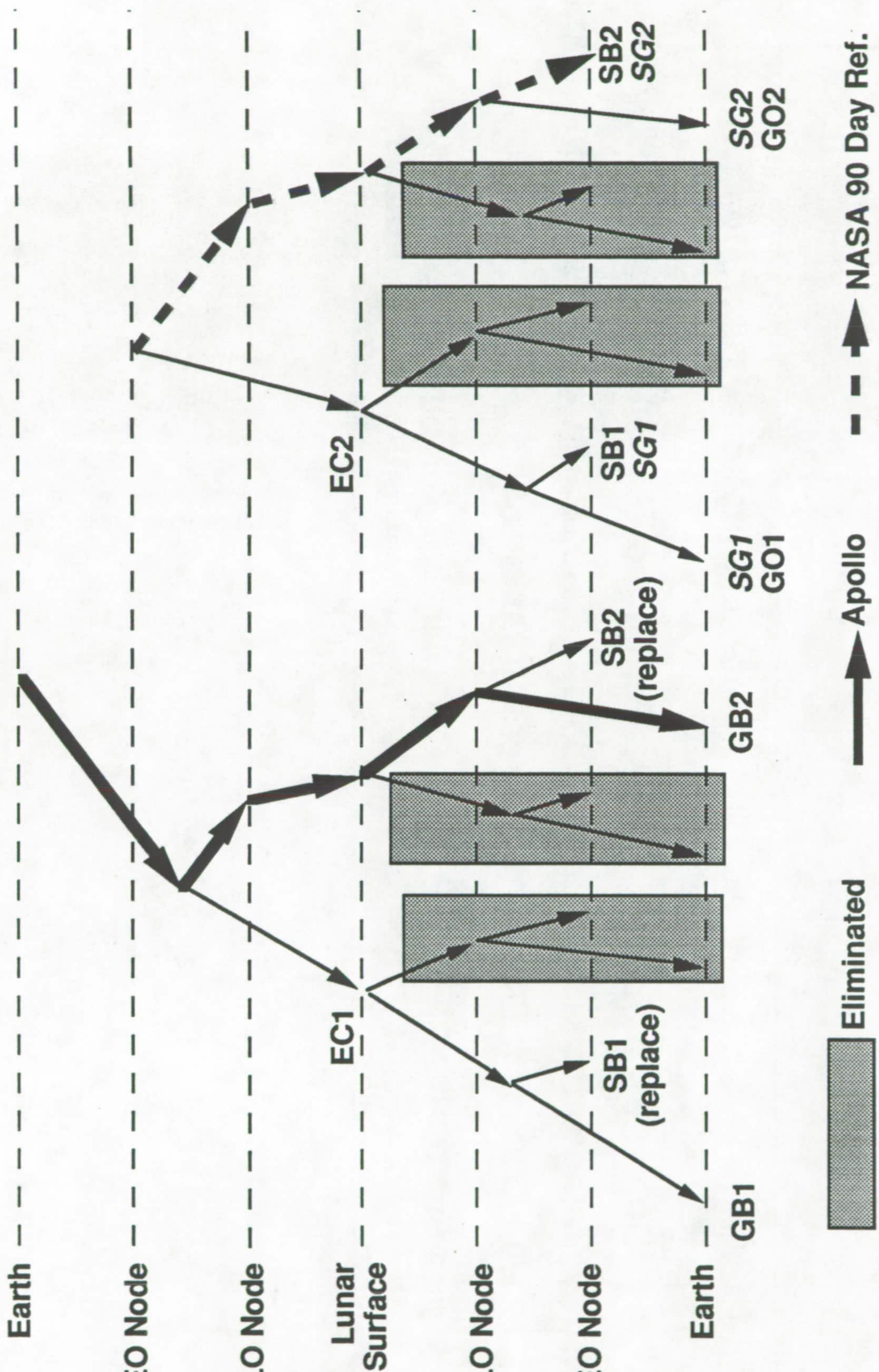


Figure 2-1.1.2.1-1. Orbital Options

SB2-2.5DP → Three cases have a P for all-propulsive return to LEO

Dual crew module (D = dual, S = single, H = hybrid)

2.5 stage (stage means propulsion + tanks, .5 stage means drop tanks)

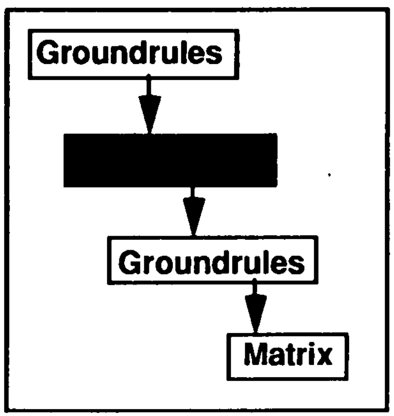
XX2 = LLO Node (LOR), XX1 = Direct to Lunar Surface

Space based, (GB = Ground Based; GO = Ground based with LEO assembly via rendezvous; SG = Space based stage, Ground based Crew module)

Figure 2-1.1.2.1-2. Mission Scenario Development

INITIAL ARCHITECTURE MATRIX SCRUBBED

- Single stage options deleted
- Expendable cargo scenarios using LLO deleted
- Crew cabs never expended
- Propulsion/tankage reusability contained in mission scenarios
- Aerobrake & re-entry return options simplified per groundrules



	GB1	GB2	SB1	SB2	SG1	SB2	GO1	GO2	EC1	EC2	EC3	EC4
	E	E	LEO	LEO	LEO	LEO	LEO	LEO	E	LEO	E	LEO
	LS	LLO	LS	LLO	LS	LLO	LS	LLO	LS	LS	LLO	LLO
	E	LLO	LEO	LLO	E/	LLO	E	LLO			LS	LS
		E		LEO	LEO	E/		E				
						LEO						
Grnd launched	X	X							X		x	
Grnd launched, LEO assy							X	X		X		x
Space based			X	X								
Comb space/grnd bsd					X	X						
All-propulsive return			X	X								
Aerobrake CC			X	X								
A/B CC + P/A together			X	X								
A/B CC + P/A separately			X	X								
Re-enter CC only	X	X			X	X	X	X				
Re-enter CC + P/A	X	X			X	X	X	X				
A/B P/A, Re-enter CC					X	X						
All prop P/A, Re-enter CC					X	X						
Expend tanks	X	X	X	X	X	X	X	X	X	X	X	X
Expend propulsion	X	X	X	X	X	X	X	X	X	X	X	X
Expend CC		X		X		X		X				
Reuse CC	X	X	X	X	X	X	X	X				
Reuse tanks			X	X	X	X						
Reuse propulsion	X	X	X	X	X	X	X	X				
1 stage	X						X		X	X		
1.5 stage	X	X	X	X	X	X	X	X	X	X	x	x
2 stage	X	X	X	X	X	X	X	X	X	X	x	x
2.5 stage	X	X	X	X	X	X	X	X			x	x
3 stage	X	X	X	X	X	X	X	X			x	x
3.5 stage		X	X	X	X	X		X				
4 stage		X		X		X		X				
1 CC	X	X	X	X	X	X	X	X				
2 CC		X		X		X		X				
Hybrid CC		X		X		X		X				

Figure 2-1.1.2.1-3. Initial Scenario Matrix

2. The minimum stage vehicle to be considered was a 1.5 stage (delete 1 stage).
3. The return options were consolidated.
4. Crew cabs were never expended, and tank and propulsion expendability or reuse is contained in the mission scenarios as will be shown later.

The resulting matrix shown in Figure 2-1.1.2.1-4 was used to generate the scenarios. For example, GB1 had a total of 4 scenarios, one for each staging option. GB2 had a total of 18 scenarios; each of the 6 staging options had 3 crew module options. Both SB2 and SG2 each had an additional scenario generated to exercise the all-propulsive versus aerobraked return to LEO trade. For each of these orbital options, two scenarios were generated for the 2.5 stage with a hybrid crew module. For the SB2 case, the final stage and crew modules were both aerobraked and all-propulsively returned to LEO in different scenarios. For the SG2 case, the crew module returns to Earth and in one scenario the final stage is aerobraked to LEO. In the other scenario, the final stage is all-propulsively returned to LEO.

Mission scenarios were developed for each of the 94 architectural options identified with the reduced matrix. For each option, an overview and timeline were developed; the mission phases and operations in which each generic flight element was involved were defined; and the characteristics and requirements for each scenario identified. Examples of these characterizations are shown in Figures 2-1.1.2.1-5 through 2-1.1.2.1-9.

2-1.1.2.2 Flight Element Definition

Using the mission scenarios, unique flight elements were identified and characterized. A functional split was made between flight elements to distinguish mass and subsystem definitions, as well as unique hardware and operations. The ultimate goal was to identify concept differences that distinguished hardware and operations costs.

The process for defining unique flight elements to support the cost assessments included a description of all vehicle options identified in the mission scenarios,

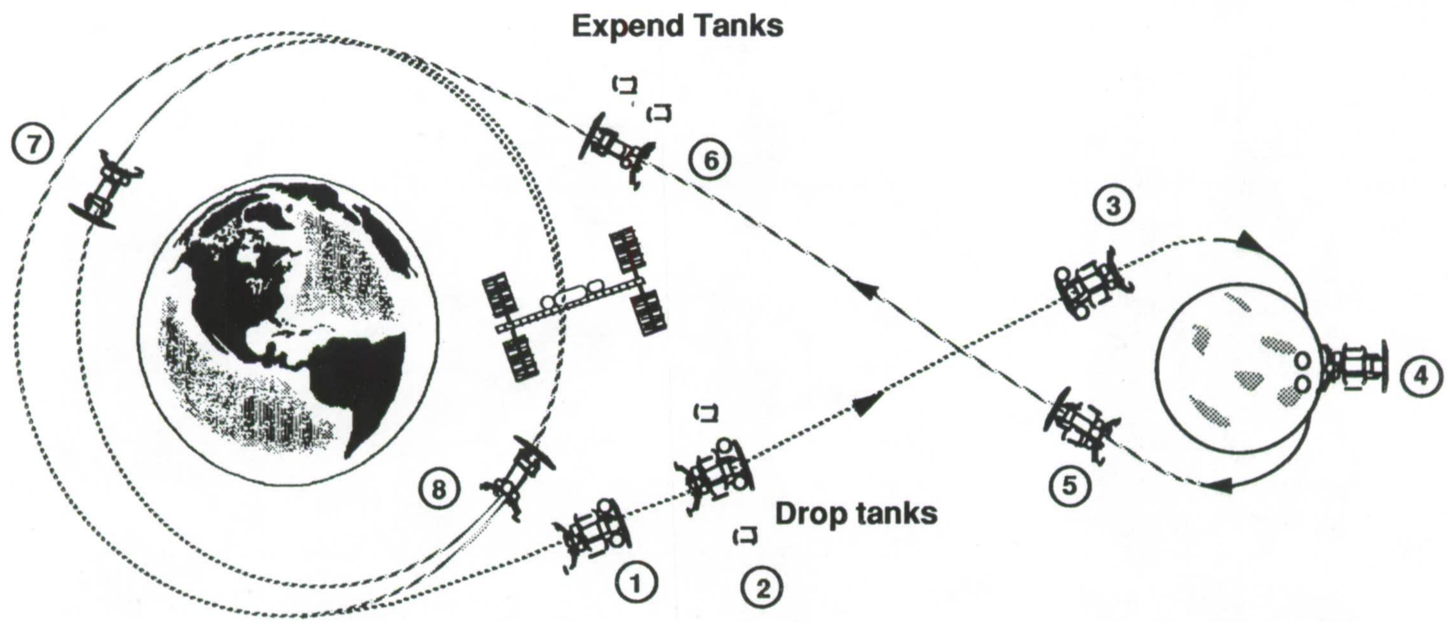
Orbital Options

		GB1	GB2	SB1	SB2	SG1	SG2	GO1	GO2	EC1	EC2
		E	*E*	*LEO*	*LEO*	*LEO*	*LEO*	*LEO*	*LEO*	*E*	*LEO*
		LS	*LLO*	*LS*	*LLO*	*LS*	*LLO*	*LS*	*LLO*	*LS*	*LS*
		E	*LLO*	*LEO*	*LLO*	*E/*	*LLO*	*E*	*LLO*		
			E		*LEO*	*LEO*	*E/*		*E*		
							LEO				
Basing	Grnd based single Inch	X	X							X	
	Grnd bsd, on-orbit assy							X	X		X
	Space based			X	■						
	Comb space/grnd bsd					X	X				
Recovery	All-propulsive return				X						
	Aerobrake return			X	■						
	Re-entry return	X	X					X	X		
	Aerobrake P/A, Re-enter CC					X	X				
	All prop P/A return, Re-enter CC						X				
Stages	1.5 stage	X	X	X	X	X	X	X	X	X	X
	2 stage	X	X	X	X	X	X	X	X	X	X
	2.5 stage	X	X	X	■	X	X	X	X		
	3 stage	X	X	X	X	X	X	X	X		
	3.5 stage		X		X		X		X		
	4 stage		X		X		X		X		
Crew Cabs	1 CC	X	X	X	X	X	X	X	X		
	2 CC		X		■		X		X		
	Hybrid CC		X		X		X		X		
	Number of scenarios -	4	18	4	19	4	19	4	18	2	2 = 94
	Trending selection -	2	9	2	12	1	8	4	4	0	0 = 42

☐ - APOLLO ■ - 90 DAY REFERENCE

Figure 2-1.1.2.1-4. Scenario Matrix

CONFIGURATION OPTION SB1-1.5



	Event	ΔV (m/s)	ΔT (hrs)
1)	MPS TLI burn	3,300	0.3
2)	ACS coast/corrections	10	84.0
3)	MPS Lunar landing	2,510	0.2
4)	Surface storage of veh	0	6 mos
5)	Lunar takeoff	2,510	0.2
6)	ACS coast/corrections	10	84.0
7)	Aerobrake maneuver		0.1
8)	Earth circularization burn	310	0.1

Figure 2-1.1.2.1-5. Mission Scenario and Timeline Example

CONFIGURATION OPTION SB2-1.5S

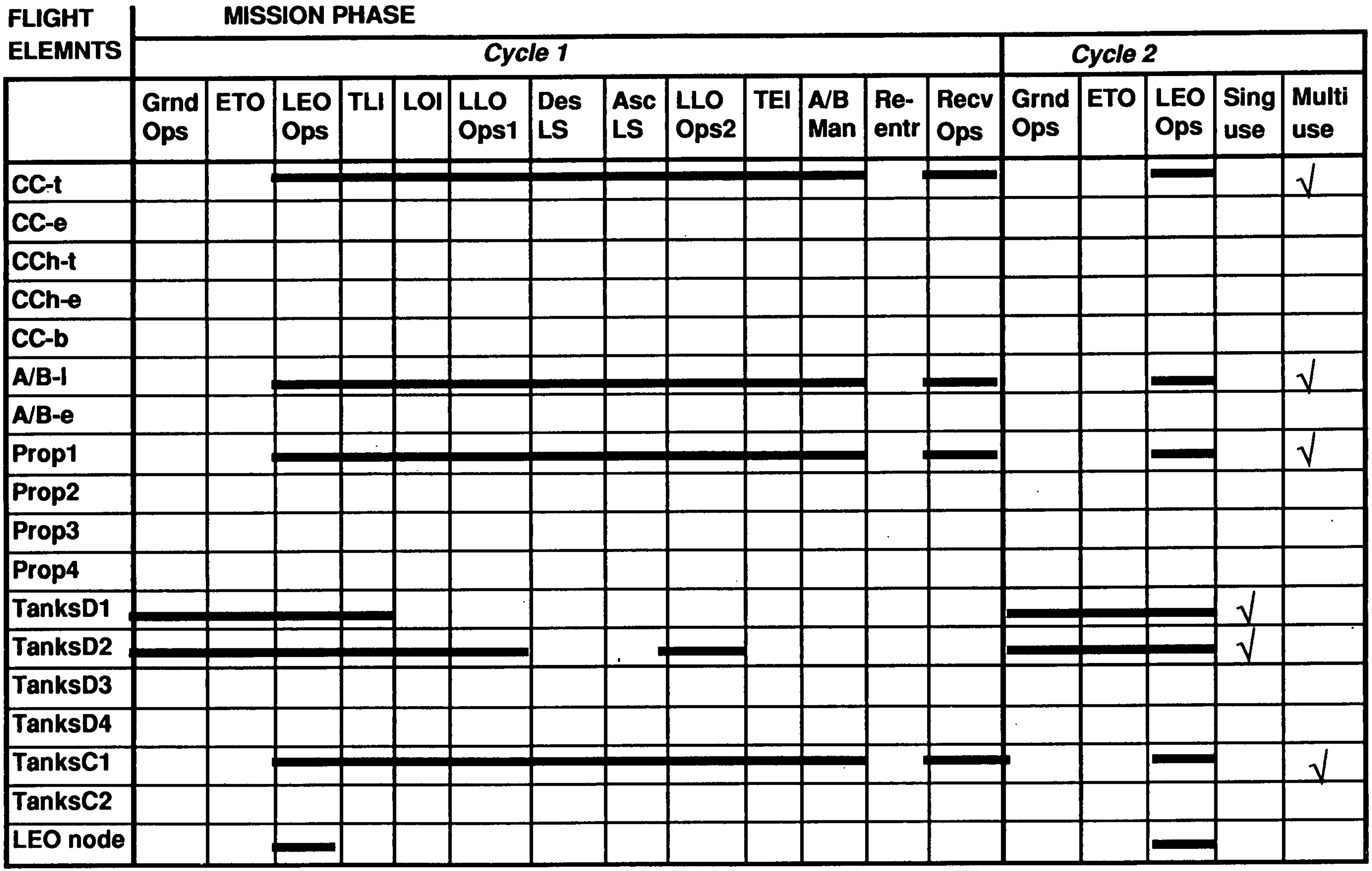


Figure 2-1.1.2.1-6. Flight Elements Per Mission Phase

CONFIGURATION OPTION SB2-1.5S

FLIGHT ELEMENTS	OPERATIONS ELEMENTS								
	Ground prcsng each flight	Ground prcsng replcmnt flt only	Ground refurb	Space refurb	Ground assy to vehicle	Space assy to vehicle	Space assy of element	Ground flight rdnss test	Space flight rdnss test
Aerobrake		√		√		√			√
CC-t		√		√		√			√
CC-e									
CCh-t									
CCh-e									
CC-b									
Prop1		√		√		√			√
Prop2									
Prop3									
Prop4									
TanksD1	√					√			√
TanksD2	√					√			√
TanksD3									
TanksD4									
TanksC1		√		√		√			√
TanksC2									
Power		√		√					√
Avionics		√		√					√
Thermal		√		√					√
Payload						√			√

Figure 2-1.1.2.1-7. Operations Per Flight Element

CONFIGURATION OPTION SB2-1.5S

FLIGHT ELEMENTS	MISSION OPERATIONS ELEMENTS: Rend & Dock in LLO	Crew Transfer in LLO	Cargo Transfer in LLO	Propellant Transfer in LLO	Storage in LLO	Storage on LS	Crew Transfer in LEO	LEO Prox Ops
Aerobrake	√				√			√
CC-t	√					√	√	√
CC-e								
CCh-t								
CCh-e								
CC-b								
Prop1	√					√		√
Prop2								
Prop3								
Prop4								
TanksD1								√
TanksD2	√			√	√			√
TanksD3								
TanksD4								
TanksC1	√			√	√	√		√
TanksC2								
Payload	√		√					√

Figure 2-1.1.2.1-8. Mission Operations Per Flight Element

CONFIGURATION OPTION SB2-1.5S

REQUIREMENTS

- Deliver 4 crew + 13.0 mt to Lunar surface once per year between 2004 and 2026
- Return crew and 500 kg cargo per flight
- Deliver in addition to the crew/cargo flights, 108 mt to the LS between 2002 and 2026
- 57 mt of cargo must be emplaced on the LS between 2002 and 2004
- Support 4 crew on the LS for up to 48 hours
- Main propulsion shall be LO2/LH

CHARACTERISTICS

- Space based, reusable
- 1.5 stage vehicle with TLI drop tanks. Single stage carries CC to and from LEO, cargo from LEO-LLO-LS, and CC from LLO-LS-LLO
- Single crew cab
- LLO ops to leave return propellant in LLO and again to rendezvous, dock and pick up same.
- Single CC, stage & core tanks aerobraked and reused for a total of 5 flights.

Figure 2-1.1.2.1-9. Scenario Requirements and Characteristics

an analysis of mission functions to identify functionally unique flight elements, and a mass definition of unique flight elements to support the cost analysis.

In parallel, mission performance of trade study options was calculated using mass trending data generated from a database of previous designs. The results of the performance analysis (see section 2-1.1.2.3) were then used to identify vehicle sizings and provide booster requirements for life cycle cost (LCC) analysis. The flight element definition process is shown in Figure 2-1.1.2.2-1. A reference vehicle was chosen to exercise the performance, mass definition, and costing process and to provide examples of the procedure. The reference chosen was the space-based, 1.5-stage vehicle with single crew cab, using the LLO node (i.e., SB2-1.5S).

In the definition of mission scenarios, flight element designators were used to identify generic flight elements in a vehicle stack. For example, CC-t was a transfer crew cab for use in the dual crew module scenario; CCh-t was a transfer crew cab for use in the hybrid crew module scenario; A/B-L was an aerobrake for LEO return; and A/B-b was an aeroshield for Earth return. These gave a functional decomposition of a generic vehicle stack, including numbers of stages and tank modules and order of propulsion unit firings. After defining the vehicles in the 94-mission architecture trade matrix, a set of descriptors was chosen to apply to unique groups of flight elements from the 536 identified flight elements. A numbered designator was used to facilitate use in the LCC model spreadsheet. Figure 2-1.1.2.2-2 shows how the elements designated in the mission scenarios were grouped by unique flight element categories and assigned numbered designators.

The general categories of flight elements included aerobrakes, transfer stages, ascent stages, lander stages, droptank modules, transfer crew modules, and excursion crew modules. For ease of costing and mass definition, propulsive units and core tank modules were combined as stages. Figure 2-1.1.2.2-3 is an example of a space-based, 3.5-stage vehicle with dual crew modules that used lunar orbit for LEV storage (SB2-3.5D). This vehicle combined flight elements from all of the categories.

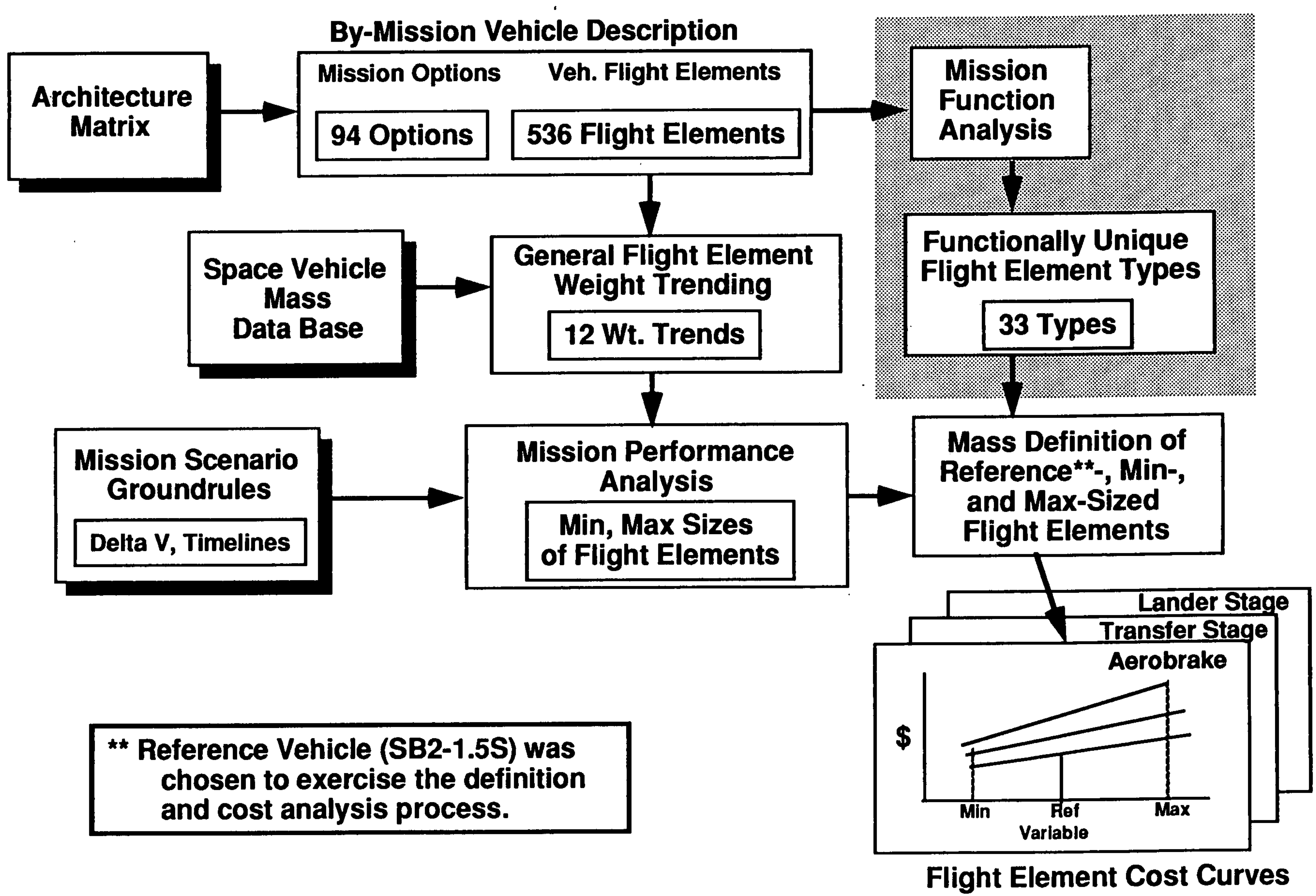


Figure 2-1.1.2.2-1. Flight Element Definition Process

Generic Vehicle Functional Decomposition

Description	Designator
Crew Module	CC-t
	CCh-t
	CC-b
	CC-e
	CCh-e
Aerobrake	A/B-L
	A/B-e
Propulsion Unit	Prop1
	Prop2
	.
	.
Reusable (Core) Tanks	TanksC1
	TanksC2
Drop-Tank Modules	TanksD1
	TanksD2
	.
	.

Trade Matrix-Related Unique Flight Element Types

Description	Designator
Transfer Crew Module	1CB,2CB,3CB,. . .
Excursion Crew Module	7CE,8CE,9CE,. . .
Aerobrake	1A,2A,3A,. . .
Transfer Stage	1PT,2PT,3PT,. . .
Lander Stage	5PL,6PL,7PL,. . .
Ascent Stage	9PLA,10PLA,. . .
Drop-Tank Modules	1T,2T,3T,. . .

Figure 2-1.1.2.2-2. Flight Element Designators

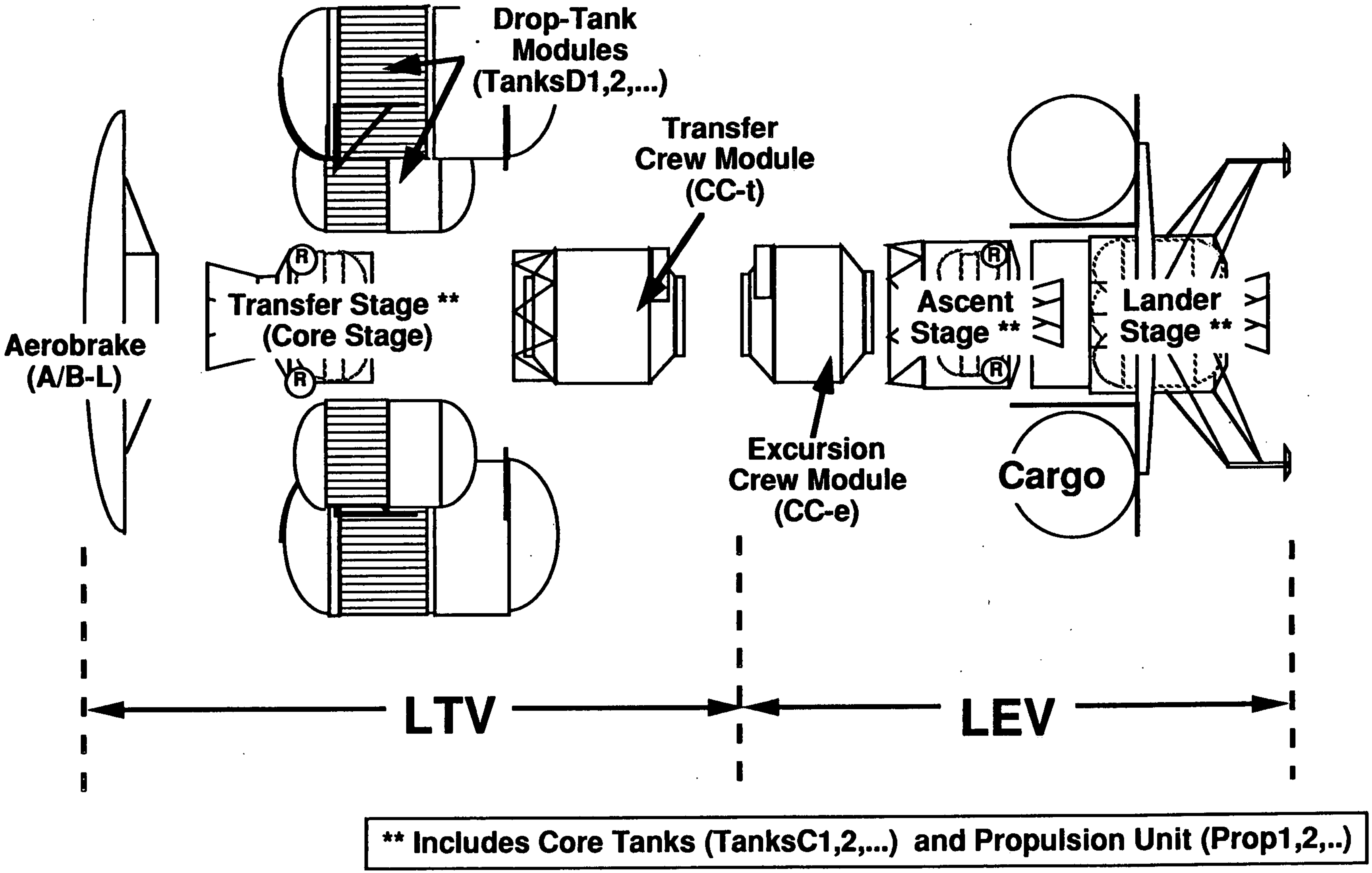


Figure 2-1.1.2.2-3. General Flight Element Categories

The SB2-1.5S vehicle, Figure 2-1.1.2.2-4, chosen as the reference case is made up of four types of flight elements: a propulsion/lander stage, single crew module, aerobrake, and droptanks. In this case, two functionally different sets of droptanks were used: one for the translunar injection (TLI) burn and the other for the remaining burns. In a steady-state piloted mode, droptanks, cargo, and crew are launched Earth to orbit and assembled with the stage, crew module, and aerobrake left in LEO. The lander tanks are filled and the TLI droptanks are expended following the TLI burn. After lunar orbit insertion (LOI), the TEI droptanks and aerobrake are left in LLO while the lander, crew module, and cargo descend to the lunar surface. Upon arrival back in LLO, the tanks and aerobrake are reattached, and the vehicle performs the trans-Earth injection (TEI) burn, after which the TEI tanksets are expended. The stage, crew module, and aerobrake perform an aeromaneuver and circularize into LEO, where they remain for refurbishment and checkout for the next flight. In the cargo missions, all flight hardware, including the lander stage and droptank sets, is launched from the ground and is flown directly to the lunar surface where it is expended.

To distinguish functionally unique flight elements, the general orbital options were analyzed on the basis of mission functions that were performed during various mission phases. Most mission-unique functions occur in LEO storage and assembly, lunar orbit operations, and Earth recovery. Mission functions that significantly affect flight hardware include rendezvous and docking, orbit stationkeeping for extended periods of time, aeromaneuvers, and Earth recovery. Mission functions that affect the above mentioned flight operations, as well as propellant transfer and storage. To identify functionally unique flight elements, the mission function analysis was applied to each mission scenario. Figure 2-1.1.2.2-5 shows the top-level functional analysis of the orbital options, while Figure 2-1.1.2.2-6 presents analysis of the reference SB2-1.5S mission scenario. In the mission scenario functional analysis, typical mission functions can be split between the participating flight elements. In the SB-1.5S case, most functions are embedded in the crew module/lander stage, but the aerobrake and tankset must remain in LLO during lunar operations, requiring independent stationkeeping capability.

Option SB2-1.5S
(Space-based with LLO node, 1.5 Stage, Single Crew Module)

	Steady-State Piloted	Cargo-only
Earth-launched Elements	TLI Drop-tanks TEI Drop-tanks Cargo Crew /Consumables	TLI Drop-tanks TEI Drop-tanks Cargo
LEO Reusable Hardware	Aerobrake Crew Module Transfer/Lander Stage	Transfer/Lander Stage
LLO Reusable Hardware	None	None
Left in LLO during Lunar Ops	Aerobrake TEI Drop-tanks	

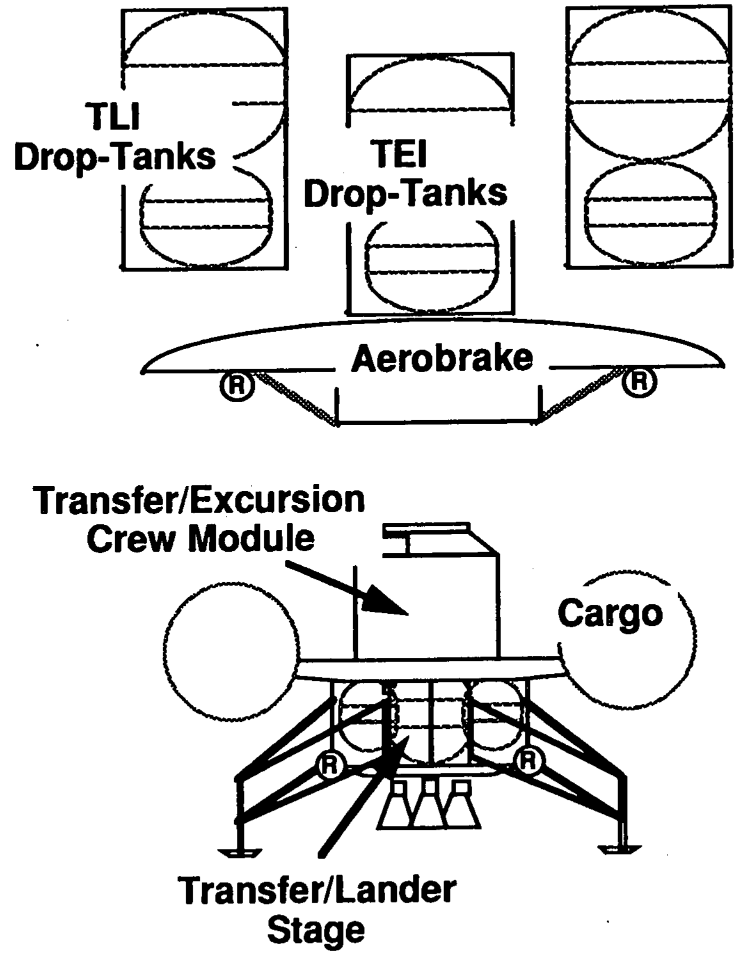


Figure 2-1.1.2.2-4. Reference Vehicle Flight Elements

Mission Phase	Mission Function	GB1	GB2	SB1	SB2	SG1	SG2	GO1	GO2	EC1	EC2
LEO Assembly	Propellant Transfer		LLO	LEO	LEO, LLO	LEO	LEO, LLO		LLO		
/ LLO Assembly	Propellant Storage		LLO	LEO	LEO, LLO	LEO	LEO, LLO	LEO	LEO, LLO		LEO
	Attitude Control		LLO	LEO	LEO, LLO	LEO	LEO, LLO	LEO	LEO, LLO		LEO
	Rendezvous / Dock		LLO	LEO	LEO, LLO	LEO	LEO, LLO	LEO	LEO, LLO		LEO
Lunar Transfer	Main Impulse	X	X	X	X	X	X	X	X	X	X
/ Earth Transfer	Attitude Control	X	X	X	X	X	X	X	X	X	X
	Life Support	X	X	X	X	X	X	X	X		
	Expend Tanksets	X	X	X	X	X	X	X	X	X	X
Lunar Descent	Main Impulse	X	X	X	X	X	X	X	X	X	X
/ Lunar Ascent	Attitude Control	X	X	X	X	X	X	X	X	X	X
	Life Support	X	X	X	X	X	X	X	X		
	Lunar Landing	X	X	X	X	X	X	X	X	X	X
Lunar Orbit	Orbit Station-keep		X		X		X		X		
Aeromaneuver	Aeromaneuver			X	X	X	X				
/ Earth Recovery	Ballistic Reentry	X	X			X	X	X	X		
	Landing / Recovery	X	X			X	X	X	X		
Between-flight	Orbit Station-keep		LLO	LEO	LEO,LLO	LEO	LEO,LLO		LLO		
Storage	Attitude Control		LLO	LEO	LEO,LLO	LEO	LEO,LLO		LLO		

Figure 2-1.1.2.2-5. Orbital Option Functional Analysis

Mission Option: SB2-1.5S

Mission Phase	Mission Function	Option SB2	Drop-Tanks TLI	Drop-Tanks TEI	Aero-Brake	Crew Module	Lander Stage
LEO Assembly	Propellant Transfer	LEO, LLO		LEO,LLO			LEO, LLO
/ LLO Assembly	Propellant Storage	LEO, LLO	LEO	LEO, LLO			LS
	Attitude Control	LEO, LLO					LEO, LLO
	Rendezvous / Dock	LEO, LLO					LEO, LLO
Lunar Transfer	Main Impulse	X					X
/ Earth Transfer	Attitude Control	X					X
	Life Support	X				X	
	Expend Tanksets	X	X	X			X
Lunar Descent	Main Impulse	X					X
/ Lunar Ascent	Attitude Control	X					X
	Life Support	X				X	
	Lunar Landing	X				X	X
Lunar Orbit	Orbit Station-keep	X		X	X		
Aeromaneuver	Aeromaneuver	X			X	X	X
/ Earth Recovery	Ballistic Reentry						
	Landing / Recovery						
Between-flight	Orbit Station-keep	LEO,LLO			X	X	X
Storage	Attitude Control	LEO,LLO					X

Figure 2-1.1.2.2-6. Reference Scenario Functional Analysis

The following discussion presents the results of the functional analysis. For each of the six major flight element types, unique flight elements are functionally described.

Transfer crew modules are used for transfer from LEO to LLO and back. Six unique transfer crew modules were identified in the mission function analysis (Figure 2-1.1.2.2-7). Crew module types 1CB, 2CB, and 3CB are ballistic return modules that return to the ground. Type 3CB is a hybrid case and returns with an excursion module attached. These crew modules are flattened biconic shapes and were chosen as a preliminary reference because of the database available from concurrent studies being done by Boeing on this type of vehicle.

Types 4CT, 5CT, and 6CT are transfer crew modules that return to, and are stored in, LEO. Type 6CT is the hybrid case and returns with an attached excursion module. Types 1CB and 4CT are used as single crew modules and go all the way to the lunar surface with the lander. The other transfer modules remain in LLO during lunar operations.

Excursion crew modules are used for lunar descent, lunar surface stay, and lunar ascent. Four unique excursion crew modules were identified (Figure 2-1.1.2.2-8) from the mission function analysis. Types 7CE and 8CE are hybrid modules; that is, they are used for the entire mission for habitable crew volume and are attached to the transfer module. During lunar operations, however, they are used as the excursion module for lunar descent and ascent. Excursion modules 9CE and 10CE are left in LLO between missions. Type 9CE must be self-sustaining in lunar orbit for mission cases in which it is the only flight hardware left in LLO.

Transfer stages provide propulsion for the transfer to and from the moon. Four unique transfer stages, shown in Figure 2-1.1.2.2-9, were identified from the mission function analysis. Types 1PT, 2PT, and 3PT are expendable stages. 1PT is expended following the TLI burn and types 2PT and 3PT are expended following TEI. Type 1PT is a "dumb" stage with controls, power, and so forth provided from another stage. 2PT and 3PT have onboard RCS for on-orbit stationkeeping. 3PT also has stationkeeping controls, avionics, and power. Type 4PT is a reusable stage that returns to LEO for between-flight storage. An

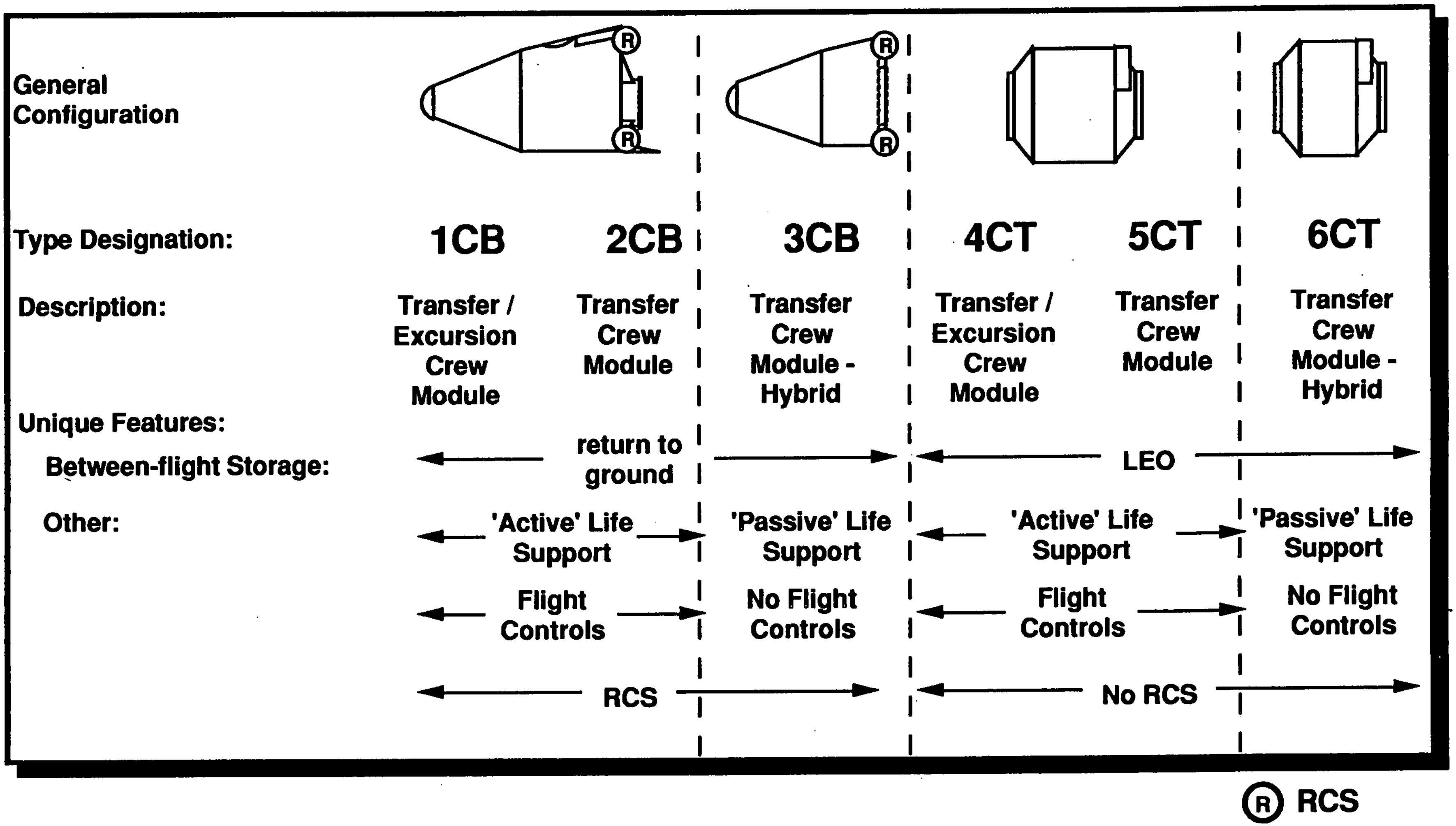


Figure 2-1.1.2.2-7. Transfer Crew Modules

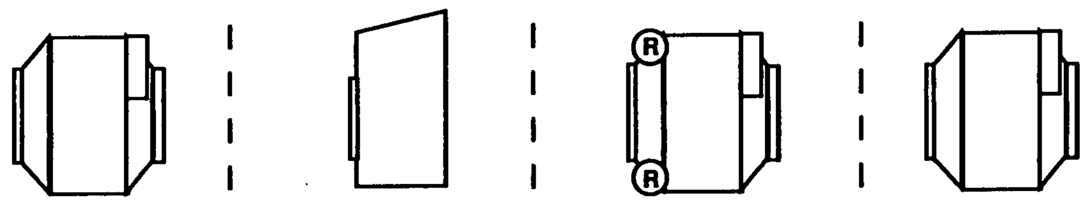
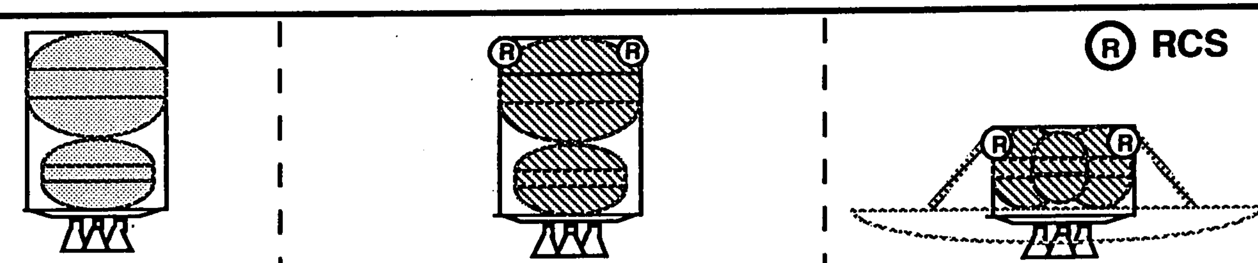
General Configuration				
Type Designation:	7CE	8CE	9CE	10CE
Description:	Excursion Crew Module - Hybrid	Exursion Crew Module - Hybrid	Excursion Crew Module - Self-sustaining	Excursion Crew Module
Unique Features:				
Between-flight Storage:	LEO	Return to ground	LLO (independent)	LLO (with lander stage)

Figure 2-1.1.2.2-8. Excursion Crew Modules

SIZE VARIES WITH PROPELLANT QUANTITY



General Configuration				
Type Designation:	1PT	2PT	3PT	4PT
Description:	TLI Stage	TEI Stage	TEI Stage	Transfer Stage
Features:				
Between-flight Storage:	expend	expend	expend	LEO
Propellant Storage Node:	LEO	←	LEO, LLO	→
Propellant Storage Time:	0 - 1 month	←	6-9 month	→
Other:	No RCS	←	RCS	→
			← Station-keep Control	→
				Aerobrake I / F
Size Range (from Performance Analysis):				
Propellant Mass (Mt):	82.2 - 142.9	30.7 - 154.9	30.7 - 183.5	0.5 - 200.3
Inert Mass (kg):	8130 - 11460	7640 - 14470	7640 - 16040	4936 - 16970

Figure 2-1.1.2.2-9. Transfer Stage Types

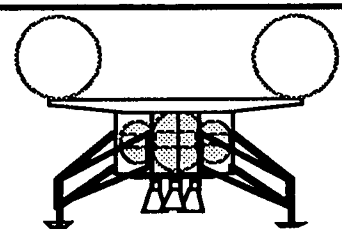
interface for attachment of the aerobrake is also provided. All transfer stage sizes vary with propellant quantity, based on mission option propellant requirements.

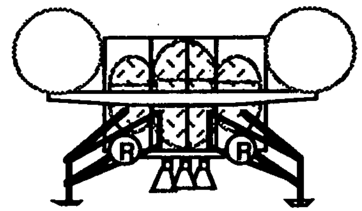
Lander stages provide propulsion for lunar desent, landing, and in some cases, ascent. Four unique lander stages, shown in Figure 2-1.1.2.2-10, were identified from the functional analysis. Types 5PLD and 6PL are expendable stages. 5PLD is left on the lunar surface, and 6PL is used for ground-based mission options and is expended following TEI. 6PL also requires RCS for attitude control during the mission. Reusable stages are identified as types 7PL and 8PL. 7PL returns to LLO where it is stored between missions, and 8PL serves as a transfer stage for space-based missions and returns to LEO. The lander stage sizes vary with propellant quantity, based on mission option propellant requirements.

Ascent stages provide propulsion for lunar ascent and in some cases, transfer back to Earth. Three unique ascent stages have been identified from the mission function analysis (Figure 2-1.1.2.2-11). Ascent stage 9PLA is expended following TLI and is the final stage for two ground-based multistage mission options. Types 10PLA and 11PLA are reusable stages. 10PLA returns to LLO where it is stored between missions, and 11PLA serves as a transfer stage for space-based missions and returns to LEO. All ascent stage sizes vary with propellant quantity, based on mission option propellant requirements.

Droptank modules, shown in Figure 2-1.1.2.2-12, are launched from the ground loaded (i.e., wet) and provide all impulse propellants for the flight vehicle. Five unique droptank modules and a propellant tanker have been identified from the mission function analysis. Droptank types 1T and 2T are expendable tank modules dropped early in the mission. Type 1T is a TLI droptank, and 2T is a lunar descent droptank. Types 3T and 4T are also expendable but are dropped later in the mission and must store propellants for longer periods of time, either on the lunar surface (4T) or in lunar orbit (3T). 4T is associated with an ascent stage, whereas 3T is associated with a transfer stage. Type 5T is a TEI droptank and is similar to 3T, except that it requires RCS and stationkeeping power and avionics for LLO storage during lunar operations. A propellant tanker for fully reusable space-based missions was defined as type 6T. The tankers are

SIZE VARIES WITH PROPELLANT QUANTITY





Ⓡ RCS

<table>
<tr><td>Type Designation:</td><td>5PLD</td><td>6PL</td><td>7PL</td><td>8PL</td></tr>
<tr><td>Features:</td><td></td><td></td><td></td><td></td></tr>
<tr><td>Between-flight Storage:</td><td colspan="2">expend</td><td>LLO</td><td>LEO</td></tr>
<tr><td>Propellant Storage Node:</td><td>LEO</td><td colspan="3">LEO, LS</td></tr>
<tr><td>Propellant Storage Time:</td><td>0 - 1 month</td><td colspan="3">1-3 month</td></tr>
<tr><td>Other:</td><td></td><td colspan="3">RCS</td></tr>
<tr><td></td><td></td><td></td><td>Station-keep Control</td><td>Aerobrake I / F</td></tr>
<tr><td>Size Range (from Performance Analysis):</td><td></td><td></td><td></td><td></td></tr>
<tr><td>Propellant Mass (Mt):</td><td>19.4 - 44.1</td><td>30.1 - 56.2</td><td>24.1 - 33.9</td><td>1.3 - 75.4</td></tr>
<tr><td>Inert Mass (kg):</td><td>5350 - 6320</td><td>5770 - 8490</td><td>7240 - 7620</td><td>6350 - 9240</td></tr>
</table>

Figure 2-1.1.2.2-10. Lander Stage Types

Flight Element Definition

SIZE VARIES WITH PROPELLANT QUANTITY

(R) RCS

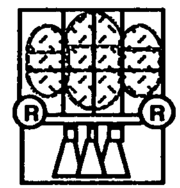
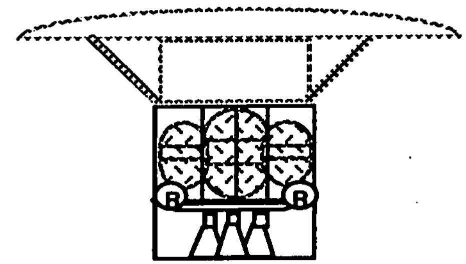
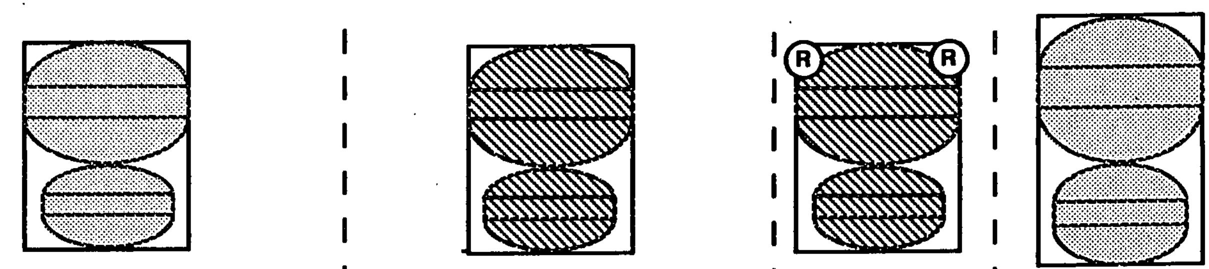
General Configuration			
Type Designation:	9PLA	10PLA	11PLA
Features:			
Between-flight Storage:	expend	LLO	LEO
Propellant Storage Node:		LEO, LS	
Propellant Storage Time:		1-3 month	
Other:		RCS	
		Station-keep Control	
			Aerobrake I / F
Size Range (from Performance Analysis):			
Propellant Mass (Mt):	15.3	5.7 - 9.6	0.5 - 1.1
Inert Mass (kg):	6800	5800 - 6430	4940 - 5040

Figure 2-1.1.2.2-11. Ascent Stage Types

SIZE VARIES WITH PROPELLANT QUANTITY

(R) RCS



General Configuration						
Type Designation:	1T	2T	3T	4T	5T	6T
Description:	TLI Drop-tank on Transfer Stage	LD Drop-tank on Lander	TEI Drop-tank on Transfer Stage	TEI Drop-tank on Ascent Stage	TEI Drop-tank on Transfer Stage	Propellant Tanker (Earth to LEO)
Features:						
Between-flight Storage:	expend					
Propellant Storage Node:	LEO		LEO, LLO	LEO, LS	LEO, LLO	LEO
Propellant Storage Time:	0-1 month		6-9 month			
Other:					RCS Station-keep Control	
Size Range (from Performance Analysis):						
Propellant Mass (Mt):	39.9 - 70.2	19.8 - 23.2	3.2 - 37.5	8.5 - 38.2	3.3	83.4 - 100.2
Inert Mass (kg):	2940 - 4610	1830 - 2020	920- 2810	1220- 2850	930	5330 - 6260

Figure 2-1.1.2.2-12. Droptank Module Types

launched to LEO and stored for filling reusable transfer stages. The droptank and tanker sizes vary with propellant quantity, based on mission option propellant requirements.

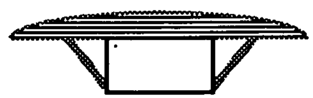
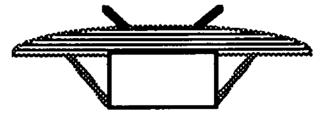
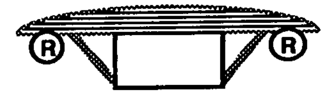
Six unique aerobrakes, used for Earth aeromaneuver to return reusable hardware to LEO, were identified. Types 1A and 2A (Figure 2-1.1.2.2-13) are similar in function in that they both go to the lunar surface and have feedline penetrations. Type 2A, however, does not need to be man rated for the aeromaneuver. 3A and 4A are similar in function in that they both have engine penetrations; however, 4A does not need to be man rated. Aerobrake types 5A and 6A both have feedline penetrations and both must have on-orbit stationkeeping capability, including power, RCS, and avionics. Type 6A does not need to be man rated.

Aerobrakes must be sized by the most stringent of three constraints: TPS peak temperature, aerodynamic stability, and wake impingement on the vehicle. In most cases, aerodynamic stability is not a sizing constraint with large symmetric aerobrakes, and wake impingement is only a problem with large, fully reusable vehicles. These constraints are also very configuration-dependent, so for the purpose of this architecture trade study the TPS peak temperature limit was used as the sizing constraint. This causes the aerobrake size to vary as a function of the total reentry mass.

An additional effort conducted as part of the flight element definition was an avionics functional and location definition. For each of the flight elements, the avionics functions associated with that flight element was identified. Figure 2-1.1.2.2-14 provides an example of the avionics functional definition for, in this case, the lander stages. This work performed in the avionics area is covered in more detail in section 1-3.5.

In summary, an analysis of the 94 mission scenarios yielded a total of 546 flight elements. Analysis of these flight elements with respect to unique mission functions resulted in 33 functionally unique flight elements. These 33 types were defined as follows:

All Rigid, Space-Assembled Aerobrakes; Size varies with Reentry Mass*

General Configuration						
Type Designation:	1A	3A	2A	4A	5A	6A
Unique Features:						
Aerobrake penetrations:	← feedlines →		← engines →		← feedlines →	
Between-flight Storage:	← LEO →					
Man-rating for Aeromaneuver:	yes	no	yes	no	yes	no
Other:	← Goes to lunar surface →				← RCS for LLO Station-keep capability →	
Size Range (from Performance Analysis):						
Brake Diameter (m):	14.1 - 14.9	9.1 - 12.4	14.0 - 14.5	9.1 - 16.8	15.4 - 15.7	11.3
Brake mass (kg):	3450 - 3840	1430 - 2680	3410 - 3660	1430 - 4910	4110 - 4300	2210

*Can also be sized to minimize wake impingement on large reusable stages (Configuration dependent).

Figure 2-1.1.2.2-13. Aerobrake Types

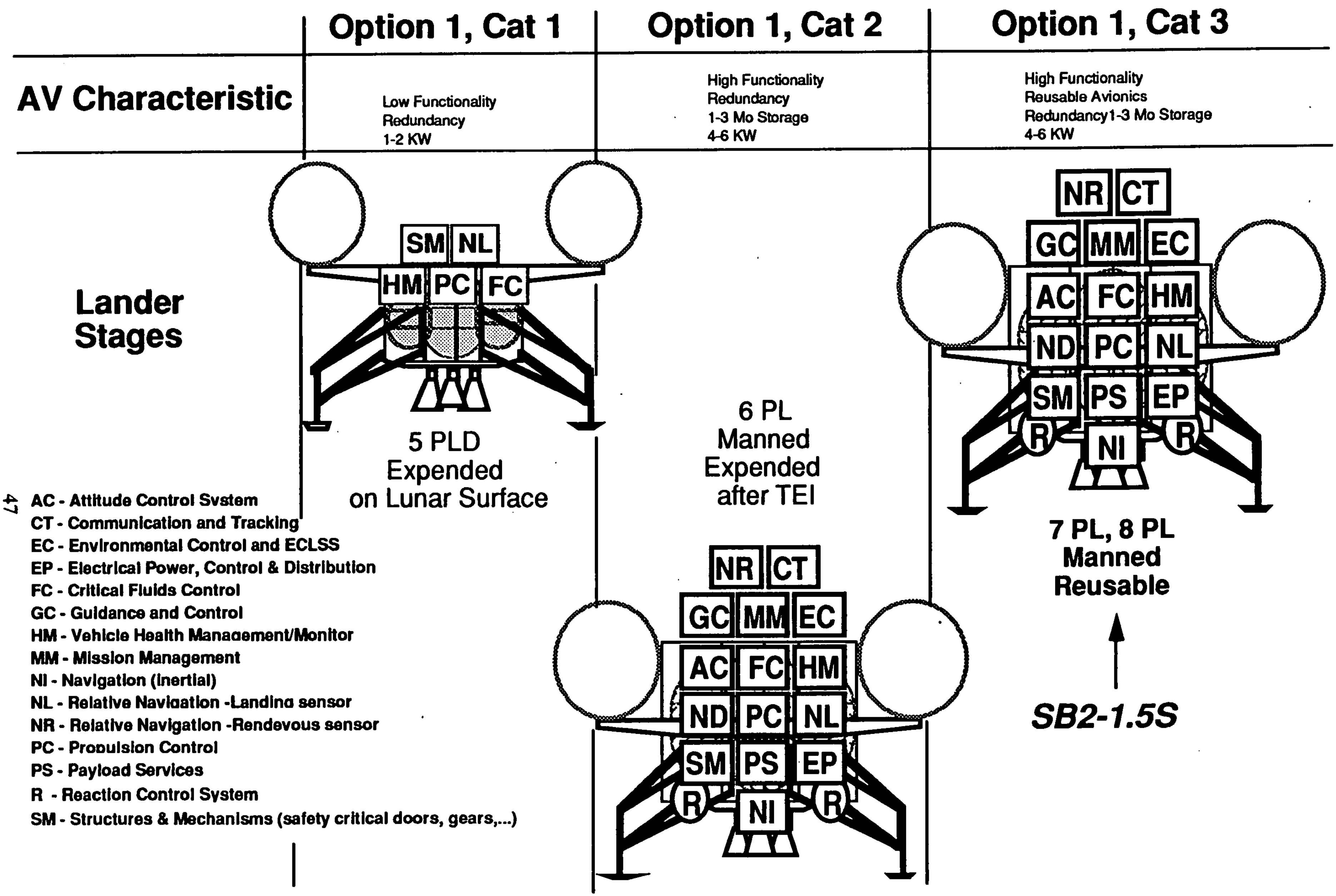


Figure 2-1.1.2.2-14. Lander Stages

1. Transfer crew modules (6 types).
2. Excursion crew modules (4).
3. Transfer stages (4).
4. Lander stages (4).
5. Ascent stages (3).
6. Droptank modules (6).
7. Aerobrakes (6).

Avionics functions required for each of these flight elements were defined to assist in the costing exercise.

2-1.1.2.3 Mission Performance Analysis

In parallel with the flight element definition analysis, mission performance of trade study options was calculated using mass trending data generated from a database of previous STV designs. The results of the performance analysis were then used to identify vehicle sizings and provide booster requirements for LCC analysis. As part of the mission performance analysis, a tank-drop optimization analysis was also conducted to determine when (i.e., after which major burns) the droptanks should be expended.

The groundrules and assumptions, weight trending equations, and mission ΔV and timelines used to produce a comparative performance analysis are contained in Figures 2-1.1.2.3-1 through 2-1.1.2.3-3. The most significant groundrule in this analysis is that performance results are not direct evaluation criteria, but they were input to cost evaluations. The analysis was designed to provide a good relative comparison between concepts as to ETO mass requirements and mass in LEO and LLO. These mass values obviously changed for the downselected vehicle designs as they were developed and optimized. However, the relative differences identified between the scenarios indicated the performance differences would remain essentially the same as any of the different scenarios were optimized.

Shown in Figure 2-1.1.2.3-4 is a mass summary and conceptual design of the reference vehicle concept SB2-1.5S, which includes subsystem masses, total dry mass, inert mass, and gross mass for each flight element as well as ETO

Note: System mass is not a direct evaluation criterion.

GROUNDRULES/ ASSUMPTIONS :

- All Cryogenic Propellant designs
- Cargo to lunar surface: 13 Mt for Steady-state Piloted; 34 Mt for Unmanned Cargo Delivery, Variable for Piloted Replacement Flight.
- Consistent Scaling Equations for all Component Sizing
- Consistent Mission Scenario delta V's and timelines
- Consistent Engine Quantity and Size (Consistent mass, thrust, Isp)
- Cargo Missions are fully Expendable with all mass launched ETO.
- Replacement flights flown with flight elements sized for Steady-state Piloted Mission.
- Cargo mission flown with flight elements sized for Piloted missions.
- No Mass penalty for Airborne Support Equipment on ETO HLLV launches
- Assumed aerobrake penetration mass penalty negligible between concepts

Figure 2-1.1.2.3-1. Performance Groundrules

CONSISTENT SCALING EQUATIONS FOR ALL COMPONENT SIZING

Crew Modules	
Ballistic Transfer Crew Module:	Wi (kg) = 10200 (inert) + 1800 (radiation protection water)
Transfer Crew Module:	Wi (kg) = 5770 (inert) + 1800 (radiation protection water)
Excursion Crew Module:	Wi (kg) = 3580 (inert)
Ballistic Hybrid Transfer Module:	Wi (kg) = 8700 (inert) + 1800 (radiation protection water)
Hybrid Transfer Module:	Wi (kg) = 2970 (inert) + 1800 (radiation protection water)
Transfer / Ascent Stages	
"Dumb" Stage:	Wi (kg) = Function of Propellant Mass
"Smart" Stage (<10Mt Wp):	Wi (kg) = Function of Propellant Mass
"Smart" Stage (> 10Mt Wp):	Wi (kg) = Function of Propellant Mass
Lander Stages	
"Dumb" Lander Stage:	Wi (kg) = Function of Propellant Mass
"Smart" Lander Stage:	Wi (kg) = Function of Propellant Mass
Drop-Tank Modules:	Wi (kg) = Function of Propellant Mass
Aerobrakes:	Wi (kg) = Function of Total Reentry Mass
Crew:	W (kg) = 800 (personnel, EVA suits, Miscellaneous)
Return Cargo:	W (kg) = 500
Flight Performance Reserve:	W (kg) = 2 percent of delta V.
Boiloff:	W (kg) = 2 percent of remaining propellant per month.
Engine Characteristics:	Isp = 481 sec (nozzle extended), 465 sec (nozzle retracted)
Delta-V Budget and Timelines per Next Chart	

Figure 2-1.1.2.3-2. Performance Analysis Criteria

LUNAR MISSION DELTA Vs:

USED IN PROPELLANT & RESIDUALS CALCULATION

Mission Event	Piloted Mission		Cargo Mission
	with Lunar Orbit	Direct to Surface	Direct to Surface
Trans-Lunar Injection	3300	3300	3300
Midcourse Correction	10	10	10
Lunar Orbit Insertion	1100	0	0
Lunar Orbit Operations	50	0	0
Lunar Descent/ Landing	2000	2510	2510
Lunar Takeoff	1900	2510	0
Trans-Earth Injection	1100	0	0
Midcourse Correction	10	10	0
Pre-Entry Correction	0	0	0
Post-Aero Circularization	310	310	0
Miscellaneous RCS	45	33	11

MISSION TIMES (Days):

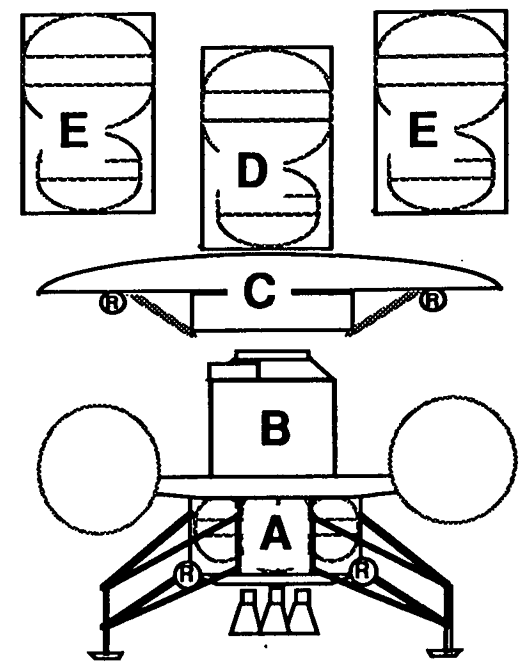
USED IN PROPELLANT BOILOFF CALCULATION

Mission Event	Piloted Mission		Cargo Mission
	with Lunar Orbit	Direct to Surface	Direct to Surface
Propellant Tanks in LEO	60	60	60
Lunar Transfer	3	3	3
Lunar Orbit Phasing	24	N/A	N/A
Lunar Orbit Stay (LTV only)	182	N/A	N/A
Lunar Surface Stay	2 (active)	2 (active)	1
Earth Transfer	3	3	N/A

Figure 2-1.1.2.3-3. Mission Delta V and Timelines

Mission Option: SB2-1.5S

Functional System	A	B	C	D	E
Structures and Mechanisms	2677	1500	1490	636	908
Tankage - Main	759	0		1082	2010
Thermal Control	378	218	1753	760	1188
Propulsion	1539		60	538	642
Power Source	173	365	90		
Power Distribution & Wiring	357	390	90		
Guidance, Nav & Control	114	248	30		
Communication & Data	125	328	65	116	204
Displays and Controls	0	91			
Environmental Control	0	740			
Personnel Provisions	0	670			
Weight Growth Margin	918	682	537	470	742
Dry Mass	**7039**	**5231**	**4115**	**3602**	**5694**
Non-propellant Consumables	0	2173	0		
Propellant - RCS, Residulas	310		25	924	1754
Inert Mass	**7348**	**7404**	**4140**	**4528**	**7450**
Crew, Suits, Cargo	13000	800			
1 Main Propellant - Usable	27700			27080	108330
Gross Mass	**48048**	**8204**	**4140**	**31608**	**115780**
2 **LEO Storage Mass**	**16385**				
ETO Mass	**191395**				
Total IMLEO	**207780**				



Notes:

All Mass in Kilograms

- **A** Transfer / Lander Stage
- **B** Transfer / Excursion Crew Module
- **C** Aerobrake
- **D** TEI Drop-Tank Module (2)
- **E** TLI Drop-Tank Module (2)

1. Lunar descent / ascent propellant launched in TEI drop-tank modules.
2. Includes lander, crew module, and aerobrake dry masses.

Figure 2-1.1.2.3-4. Reference Vehicle Mass Summary

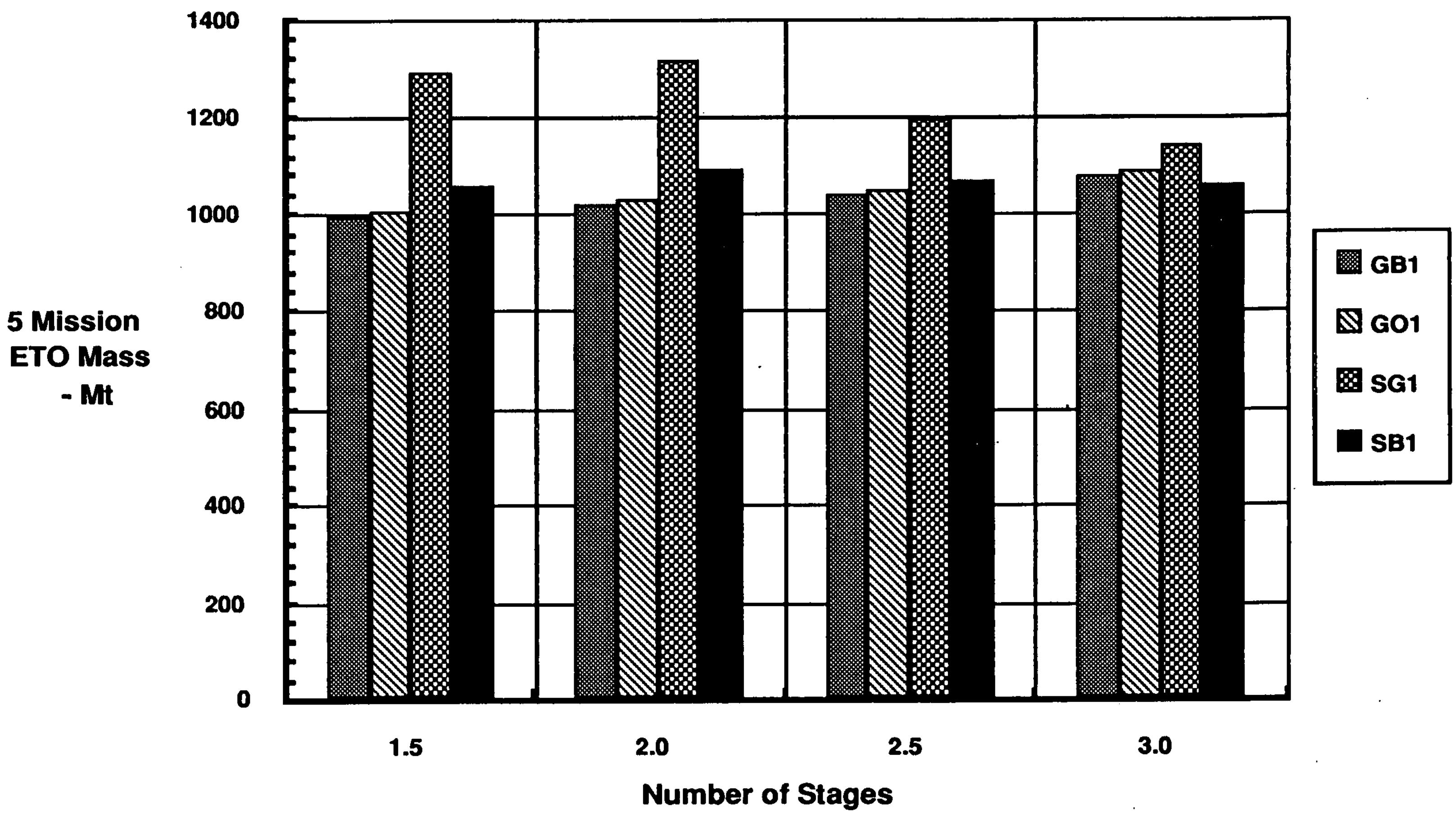
mass and IMLEO. Generally, most avionics functions are included in the crew module, as well as life support and crew provision equipment. The aerobrake includes power, avionics, and RCS hardware for stationkeeping capability in LLO during lunar operations. The propulsion stage functions as both transfer stage and lander and has onboard batteries for thrust vector control power. Other vehicle power is supplied by the crew module. The crew module also includes 1,800 kg of radiation protection water.

Comparative performance for this trade study was based on the combined ETO mass required for five piloted flights of the vehicle option as well as ETO mass for cargo delivery missions. The five piloted flights include four steady-state flights and one hardware replacement flight. Some of the results from a performance standpoint are discussed in the following paragraphs.

For single crew module designs that go directly to the lunar surface, the lowest five-flight ETO mass, shown in Figure 2-1.1.2.3-5, is the ground-based 1.5-stage vehicles. The worst cases are the combination space- and ground-based options, with 5% to 30% heavier mass than other options. These are poorer performers because both the stage aerobrake and crew module heat shield go all the way to the lunar surface. The space-based options are also poor because stage aerobrakes go to the lunar surface.

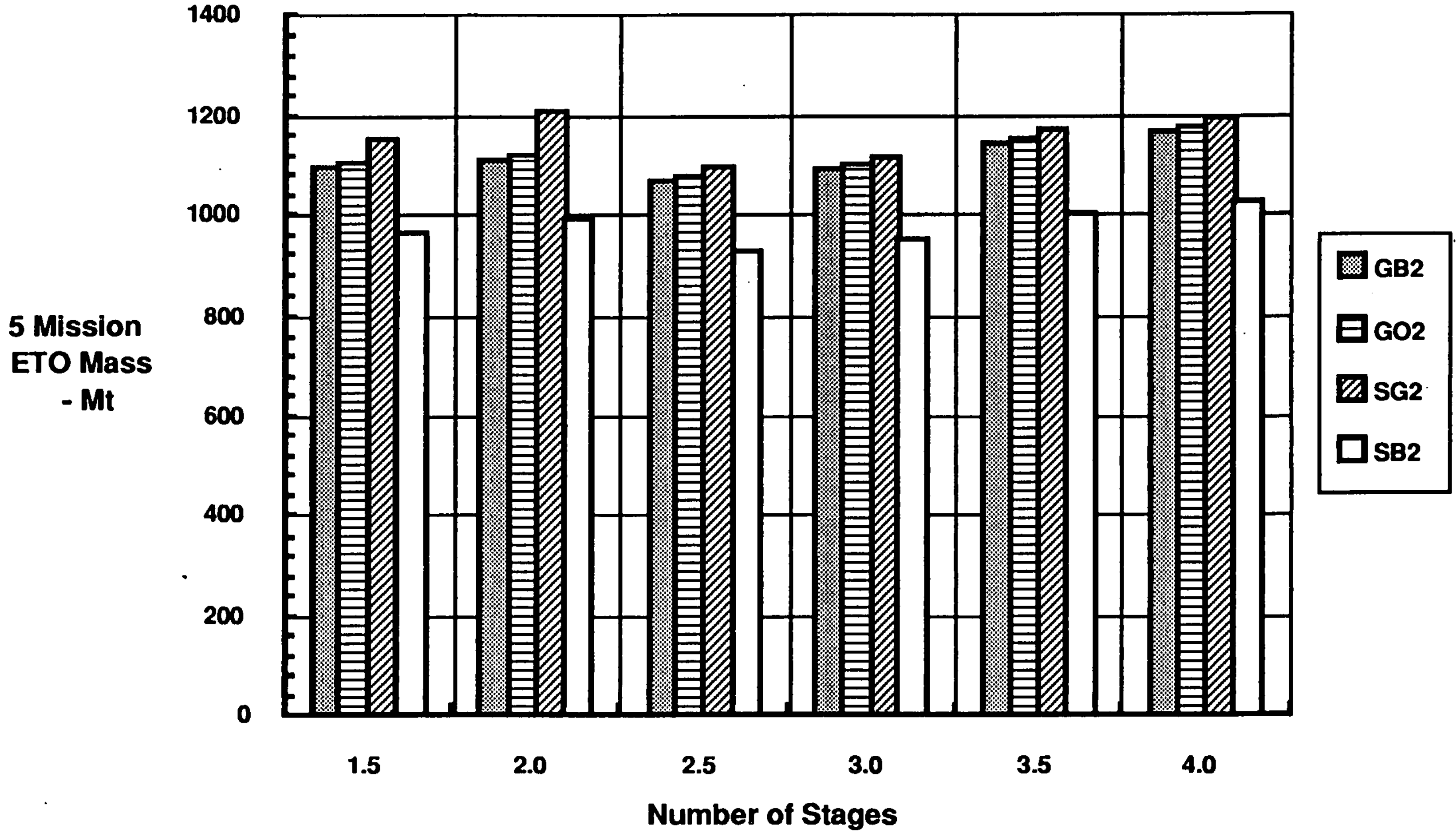
For single crew module designs that use LLO for hardware storage, the lowest five-flight ETO mass is the space-based 2.5-stage vehicles shown in Figure 2-1.1.2.3-6. These vehicles have a reusable LEV in lunar orbit and relatively lightweight transfer crew modules. The worst cases are again the combination space- and ground-based options, because of a heavier crew module (ballistic return) taken to the lunar surface. The ground-based options also have the heavier crew module, but benefit from not having aerobrakes.

Figure 2-1.1.2.3-7 shows the results for dual crew module designs that use LLO for hardware storage, with the lowest five-flight ETO mass again being seen in the space-based 2.5-stage vehicles. Again the combination space- and ground-based options are the poorest performers, because of the heavier transfer crew module. Similarly, the ground-based options have the heavier crew module but benefit from not having aerobrakes. The dual crew module cases generally are



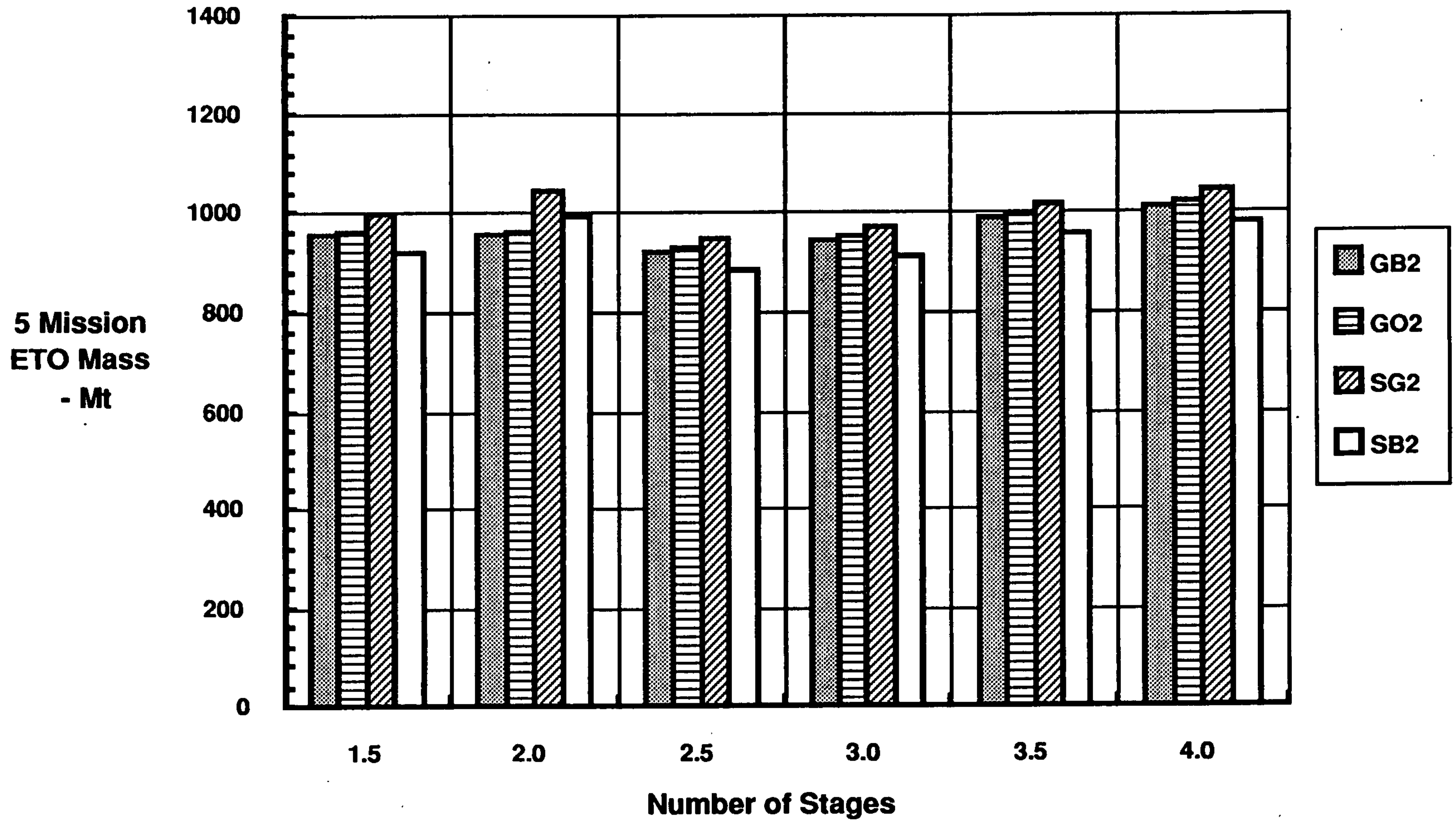
Combination space-ground basing options (SG1) are 5%-30% heavier because both aerobrake and reentry shield are taken all the way to the lunar surface

Figure 2-1.1.2.3-5. ETO Mass - Single Crew Module (LS Direct)



Space-Based option (SB2) averages 10% - 15% lighter ETO mass.

Figure 2-1.1.2.3-6. ETO Mass - Single Crew Module (LLO Node)



All basing option ETO mass values are within 5% to 8% of each other.

Figure 2-1.1.2.3-7. ETO Mass - Dual Crew Module (LLO Node)

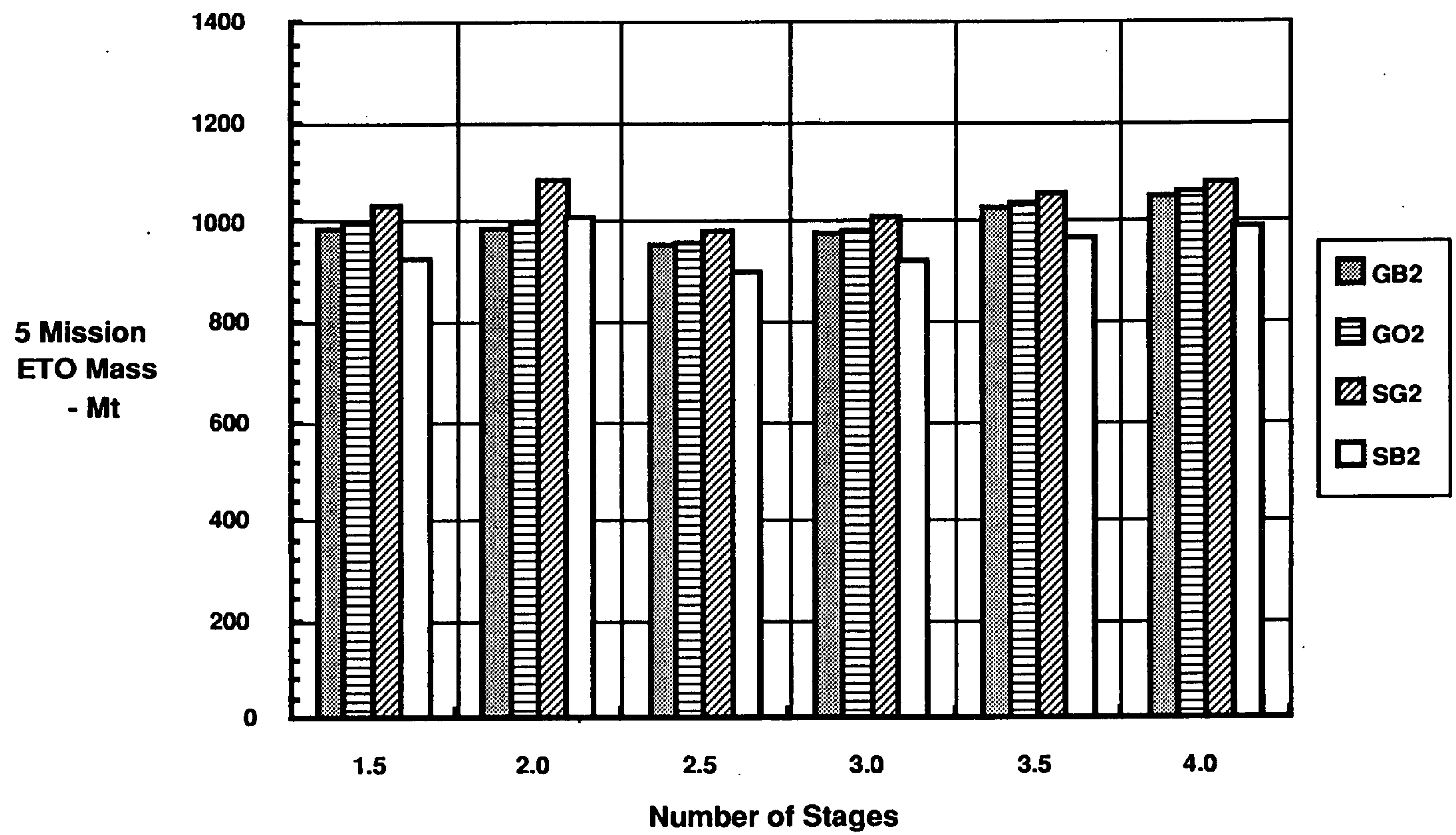
13% to 15% lighter than the corresponding single crew module cases. A significant conclusion that can be drawn within the dual crew module option data is that all ETO mass values are within 5% to 8% of each other. Thus, the dual crew module scenarios are not as performance sensitive to basing (and related configuration) impacts as are the single crew module scenarios.

The same trends that applied to the dual crew module cases apply to the hybrid crew module cases (Figure 2-1.1.2.3-8). The hybrid crew module ETO masses are 2% to 5% higher than the corresponding dual crew module masses but are 10% to 11% less than the corresponding single crew module cases. Again, the hybrid crew module scenarios are not as performance sensitive to basing (and related configuration) impacts as are the single crew module scenarios.

One of the architecture trade studies was the impact of an all-propulsive as opposed to an aeroassisted Earth-orbit insertion. For the two cases run, the all-propulsive option required 13% to 30% more ETO mass (Figure 2-1.1.2.3-9). The combination space- and ground-based case has less difference between aerobraked and all-propulsive because the crew module is not aerobraked with the vehicle, lowering the aerobraked mass significantly.

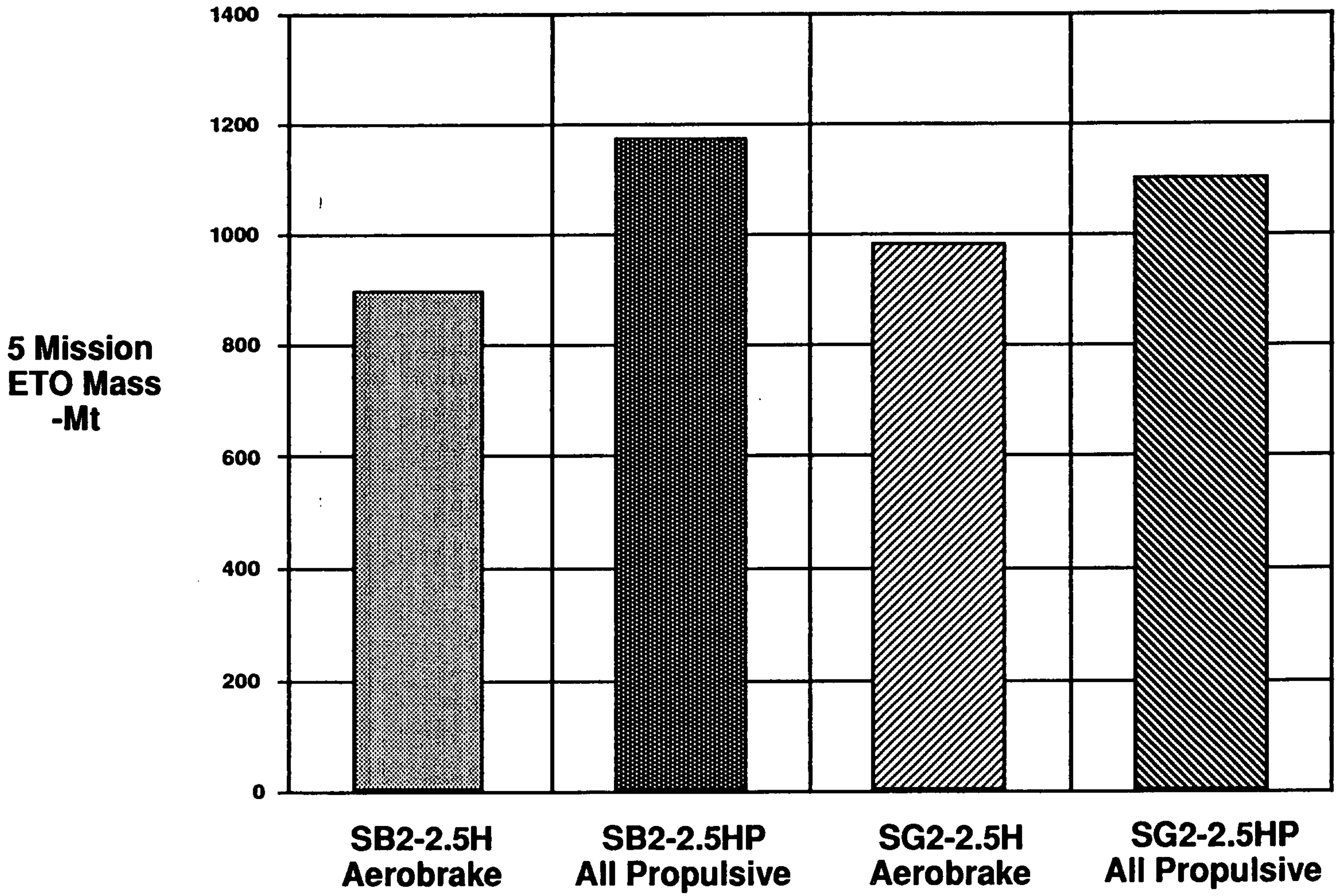
Tank-Drop Optimization Analysis. As part of the trade study analysis, an optimization of tank-drop event numbers and location was performed for 1.5-stage (direct to lunar surface) and 2.5-stage (LLO node) vehicles to check initial assumptions made in the mission scenarios and to provide a basis for future tank-drop assumptions. The analysis was performed for both space-based and ground-based options, using single crew modules for the direct to lunar surface cases and dual crew modules for the LLO node cases. Figure 2-1.1.2.3-10 provides an overview of the cases for which this analysis was performed. For each case, all combinations of tank-drop events following major burns were examined, including no tank-drop events. For LOI droptanks, it was assumed that the droptanks would not be disposed of until after rendezvous with the lunar excursion vehicle following the lunar surface operations.

Droptank disposal can occur with TLI and TEI droptanks disposed of by reentry into the atmosphere or by being boosted out of the Earth-Moon system. The latter option is accomplished prior to midcourse correction and is the preferred



All basing option ETO mass values are within 5% to 8% of each other.

Figure 2-1.1.2.3-8. ETO Mass - Hybrid Crew Module (LLO Node)



All Propulsive option averages 13% - 30% heavier ETO mass.

Figure 2-1.1.2.3-9. ETO Mass - Aerobrake Versus All-Propulsive

Goal: Find Optimum Number and Location of Tank-Drop Events:

Basing Mode:	Space-Based		Ground-based	
Mission Type:	LS Direct	LLO Node	LS Direct	LLO Node
Propulsive Stages	LTV	LTV, LEV	LTV	LTV, LEV
Crew Modules	Single	Dual	Single - Ballistic	Dual - Ballistic
No. Major Burns	4	6	3	5
No. Drop-Tank Events	0 - 3	0 - 5	0 - 2	0 - 4
No. Drop-Tank Combinations	8	32	4	16

Drop-Tank Disposal — Denotes Analysis Assumptions

Trans-lunar Injection (TLI) Tanksets:	Earth Reenter or <u>Boost Out of Earth-Moon System</u>
Lunar orbit Insertion (LOI) Tanksets:	<u>Lunar Surface Disposal</u>
Lunar Descent (LD) Tanksets:	<u>Lunar Surface Disposal</u>
Lunar Ascent (LA) Tanksets:	<u>Lunar Surface Disposal</u>
Trans-Earth Injection (TEI) Tanksets:	Earth Reenter or <u>Boost Out of Earth-Moon System</u>

Figure 2-1.1.2.3-10. Tank-Drop Optimization Analysis

option. For LOI, LD, or LA droptanks, lunar surface disposal is the method of disposal.

For the 60 tank-drop cases run, the minimum cases are plotted on Figure 2-1.1.2.3-11 as total vehicle IMLEO versus number of tank-drop events. For space-based missions, the lowest mass occurs with tank-drop events following the first and second burns (TLI and LOI for options using LOR, and TLI and lunar descent for lunar direct options). The ground-based minimum occurs with only one tank-drop event following TLI for either the LOR or lunar direct scenarios.

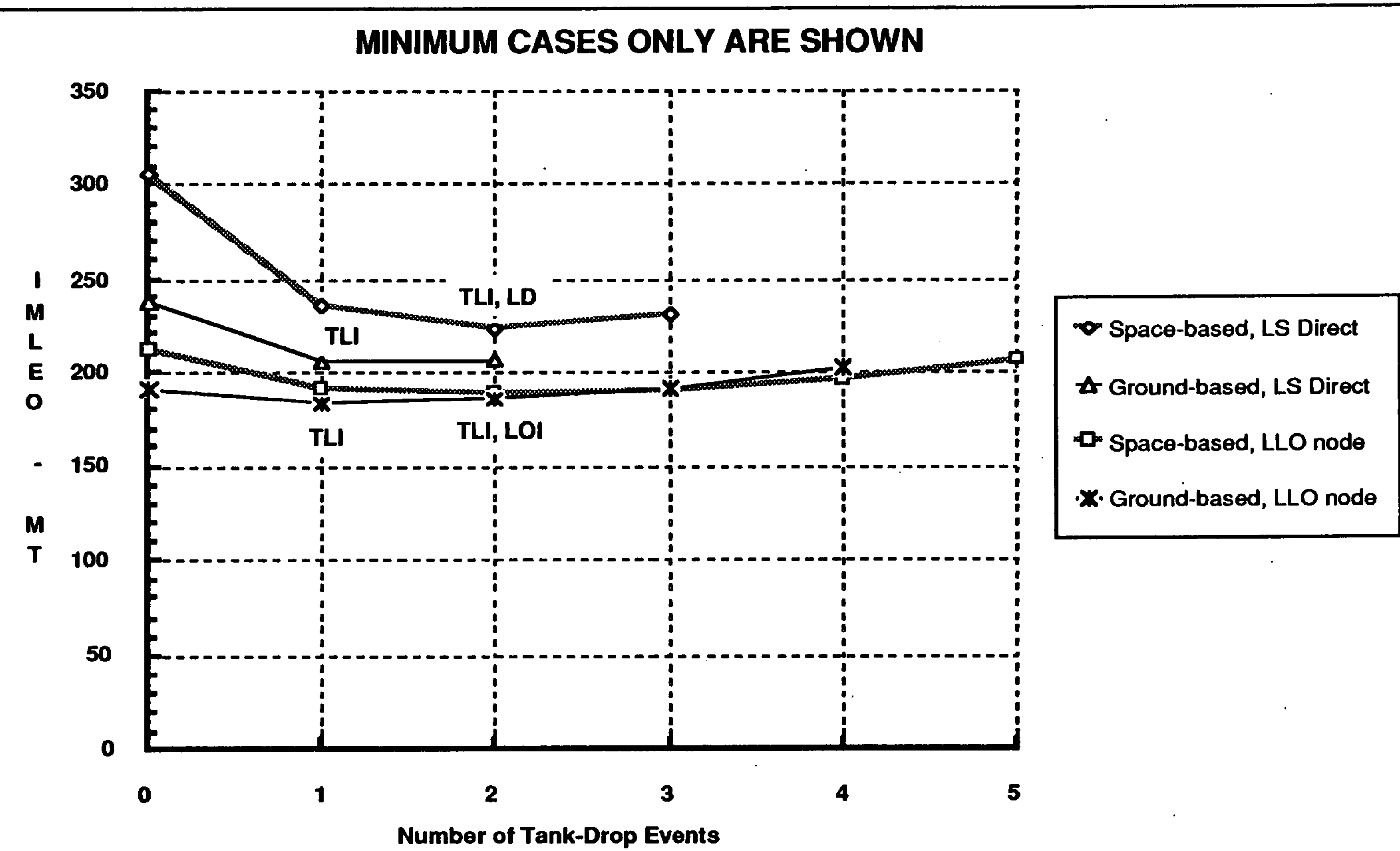
Sensitivities to the minimum cases are shown in Figure 2-1.1.2.3-12. The tank-drop cases within 5% of the minimum IMLEO cases are shown, as well as the worst cases for each basing option. High penalties occur for no droptanks on direct-to-surface vehicles and for lunar ascent droptanks on LLO node vehicles.

Also indicated are the trade study tank-drop assumptions used in the mission scenarios. The penalty for dropping tanksets following TEI for the space-based cases (assumed in the space-based mission scenarios instead of lunar descent or LOI droptanks) is within 6% of the minimum. As performance was not a direct evaluation criterion, but contributed only to LCC, the assumption of TEI instead of LD or LOI droptanks (when used in mission scenarios) was seen as insignificant in terms of relative assessments of the scenarios.

Note that when the downselected ground-based scenarios were further defined and optimized, TLI and lunar descent droptanks were used. In the more detailed design process, landing legs were left on the lunar surface. This change in staging resulted in the optimum choice for the ground-based options being the use of TLI and lunar descent droptanks instead of just TLI droptanks.

2-1.1.2.4 Operations Elements Definition

To support the cost and margins and risk assessments, and the subsystem design task, operations flows were developed for the mission scenarios. Operations were defined from the start of KSC processing of a new vehicle to the end of the mission of its second flight. This covers all major events,



Space-based missions have lowest mass with tank-drop events after the first (TLI) and second (LOI or LD) burns.

Ground-based missions have lowest mass with one tank-drop event after the first (TLI) burn.

Figure 2-1.1.2.3-11. Tank-Drop Optimization Results

Tank-Drop Cases Within 5 Percent of Minimum IMLEO

Space-based, LS Direct		Ground-based, LS Direct		Space-based, LLO Node		Ground-based, LLO Node	
Tank-Drop Events	IMLEO	Tank-Drop Events	IMLEO	Tank-Drop Events	IMLEO	Tank-Drop Events	IMLEO
TLI, LD	222.3	TLI	205.7	TLI, LOI	188.7	TLI	183.9
TLI, LD, TEI	230.7	TLI,LD	206.6	TLI, TEI	188.9	TLI, LOI	186.0
				TLI, LOI, TEI	190.9	LOI	187.5
TLI, TEI	236.2			LOI	191.4	TLI, LD	188.7
	(+6%)			LOI, TEI	192.5	TLI, LOI, LD	191.6
				TLI, LD, TEI	193.9	NONE	192.1
				TLI, LOI, LD	194.3		
				TLI	196.1		
				TLI, LD	196.1		
				TLI, LOI, LD, TEI	196.4		
				LOI,LD	196.8		
				TEI	197.3		

Minimum Case

Mission Scenario Assumption

Worst Case Tank-Drop Events

No Drop-Tanks	305.7 (+38%)	No Drop-Tanks	238.2 (+16%)	LA	248.8 (+32%)	LA, LOI	231.3 (+26%)

IMLEO performance penalty for dropping tanksets after TEI vs. after LD or LOI is insignificant for the space-based vehicles.

Figure 2-1.1.2.3-12. Tank-Drop Location Sensitivity

excepting final disposal, in the vehicle's life, including refurbishment for reflight. Figure 2-1.1.2.4-1 shows the operations element definition process.

A diverse source of inputs was considered in developing the operations flows. Studies have been performed in the past by several major contractors whose primary purpose was to define on-orbit operations of an OTV (STV of lunar vehicle). Operations were defined at a major task description level, with a ROM estimate of task duration hours assigned. Figure 2-1.1.2.4-2 demonstrates the difference in complexity between space-based and ground based scenarios. The number of operations steps required was considered as a minus in the risks and margins analysis task.

Figure 2-1.1.2.4-3 shows the top-level description of the operations defined for SB2-1.5S (space-based, 1.5-stage, single crew cab, equipment staged in LLO). Each box with "dog ears" is called a super task. A super task is a task that has another file underneath it that defines that task in detail. The times shown are task duration hours and do not represent man-hours. To perform the trade studies, the man-hours were estimated and totaled by hand, in a format more usable by the trade team. The top-level description of the operations defined for GB2-1.5S (ground-based, 1.5-stage, single crew cab, equipment staged in LLO) is shown in Figure 2-1.1.2.4-4.

By comparing space based to ground based, ground based has approximately 50% (16 out of 30) less steps performed before the start of the lunar mission. This can be looked at two ways. It implies that there is less risk in a ground-based system because there are less tasks to be performed. The other observation is that the decision to start the lunar mission for a ground-based vehicle is made before boost to LEO, where as for a space-based vehicle, it is made after. This is significant because the ETO acoustic and dynamic environment is predicted to be the worst the lunar vehicle will experience.

Figure 2-1.1.2.4-5 demonstrates what is contained in the subflows under the super task. This method of defining operations gives the other study team members a single place to record comments and design notes while reviewing the operations steps.

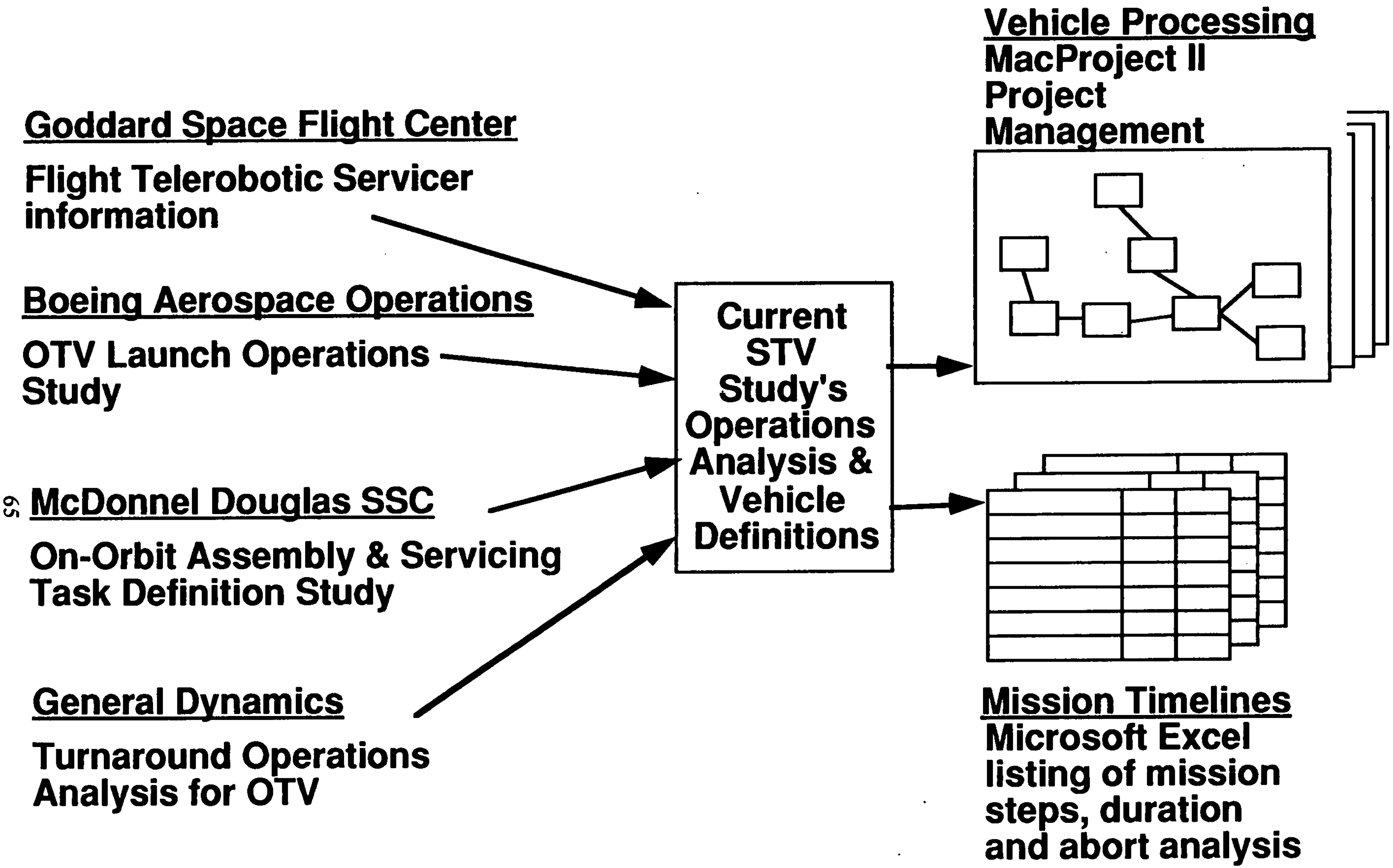


Figure 2-1.1.2.4-1. Operations Definition Process

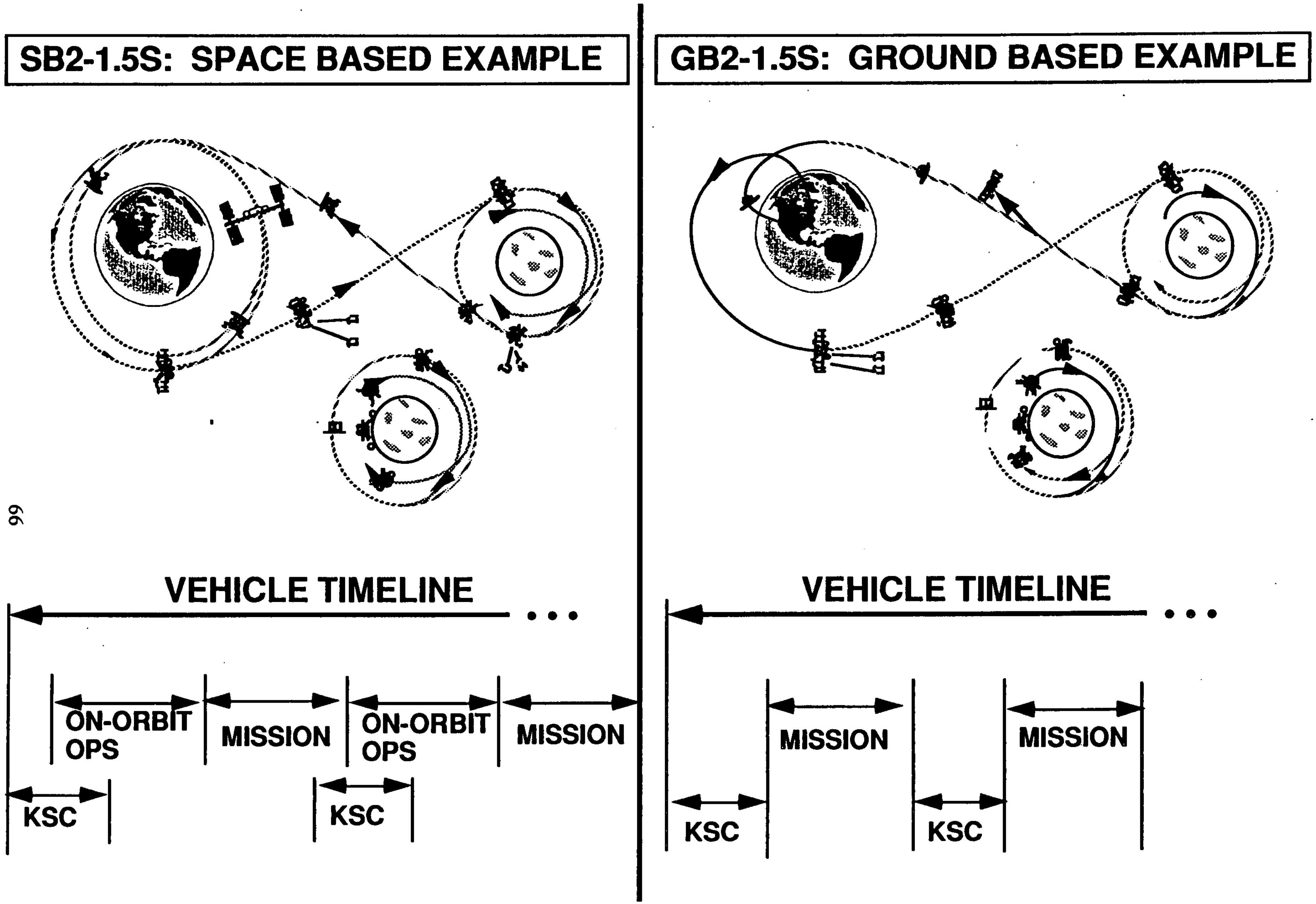
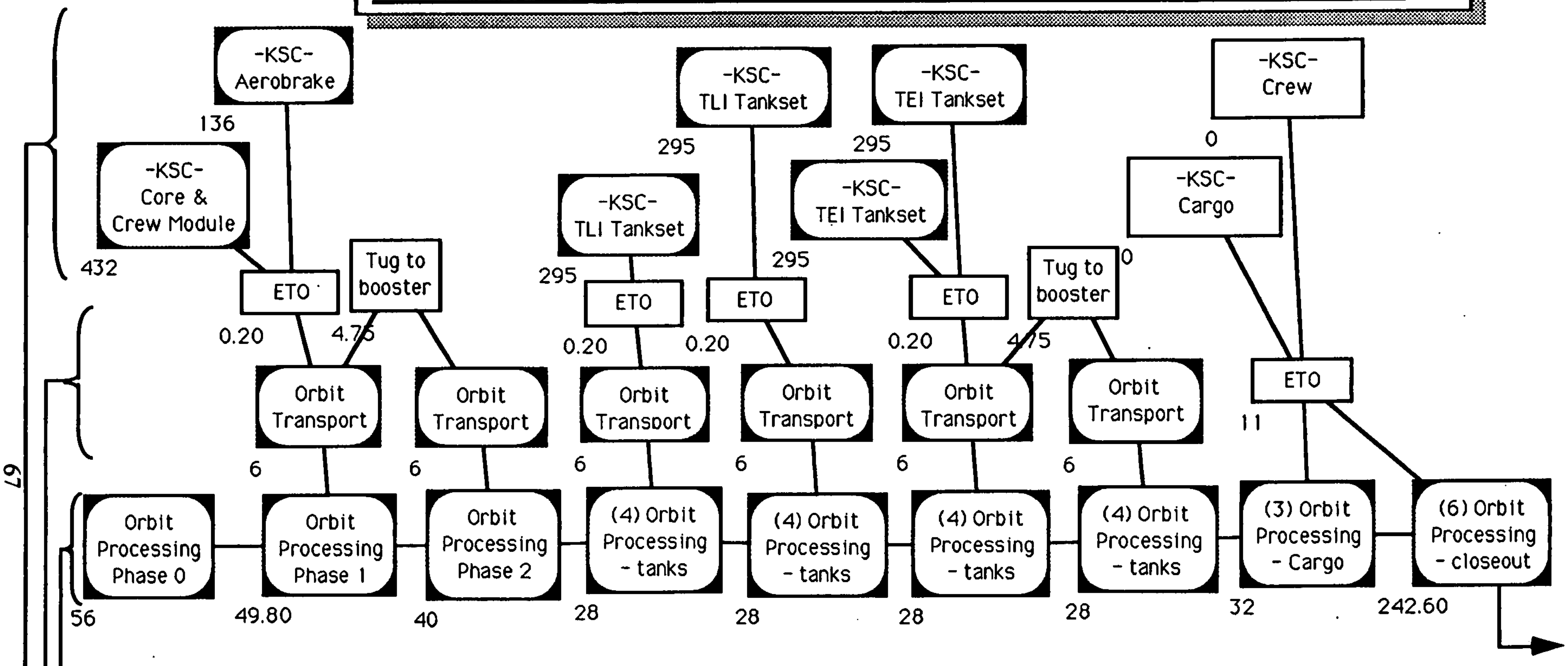


Figure 2-1.1.2.4-2. Basing Impacts on Operations Complexity

DELIVERY OF NEW SPACE BASED VEHICLE TO BASE



- Vehicle assembly at base (assumed to be SSF or equivalent)
- Vehicle element transport from KSC to base
- Vehicle element processing at KSC

Figure 2-1.1.2.4-3. Example Operations Flow - SB2-1.5S (Sheet 1 of 3)

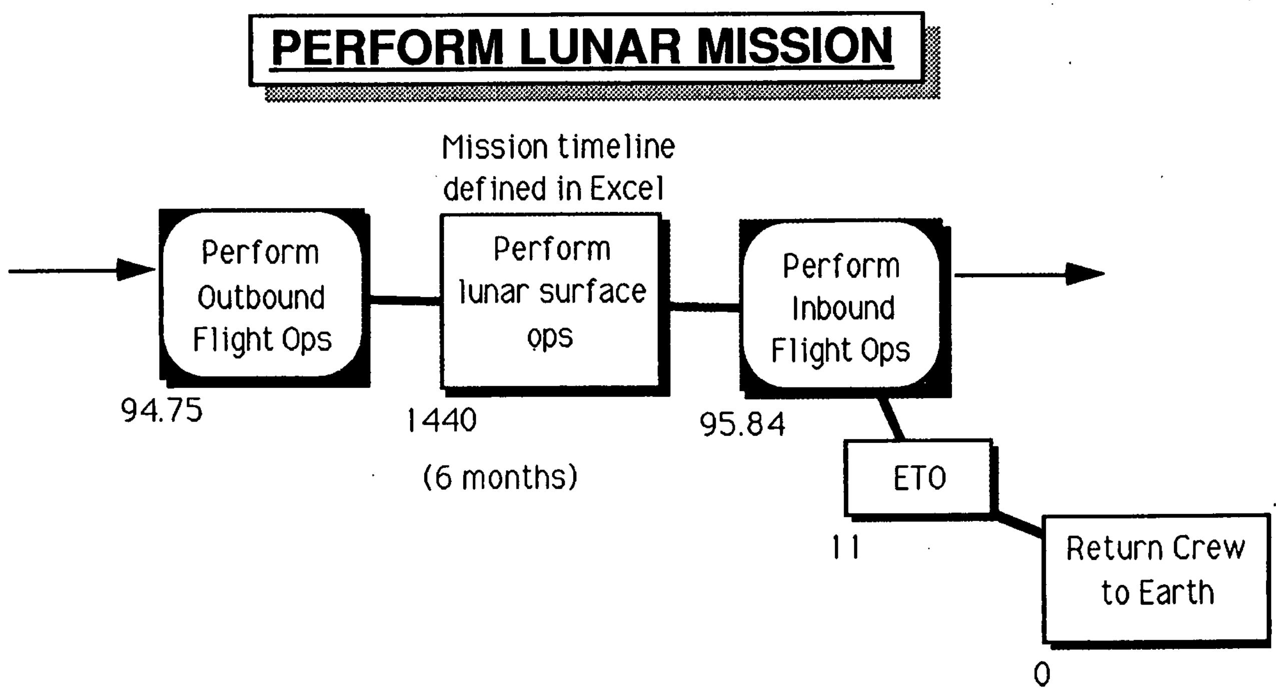
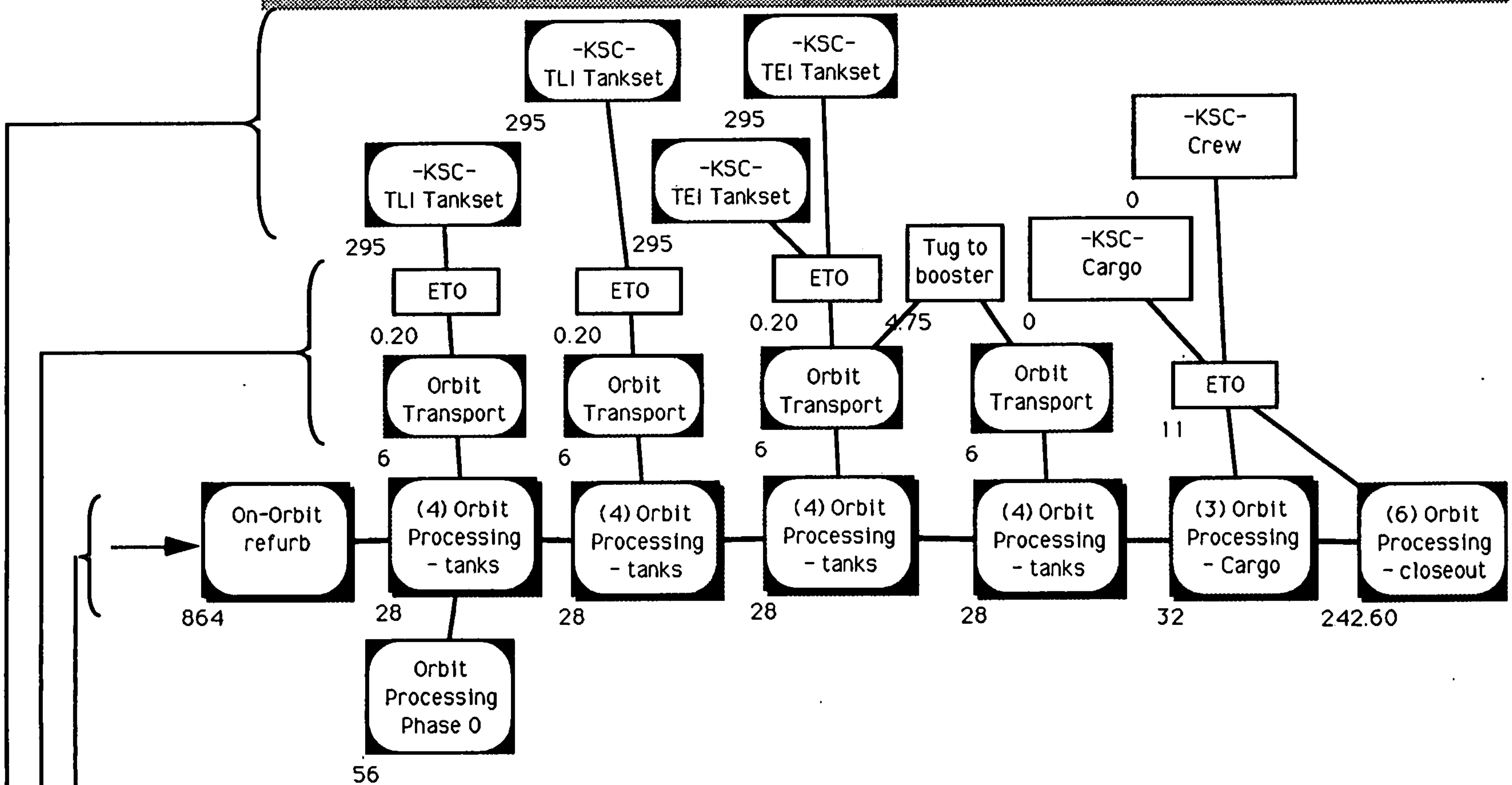


Figure 2-1.1.2.4-3. ***Example Operations Flow - SB2-1.5S (Sheet 2 of 3)***

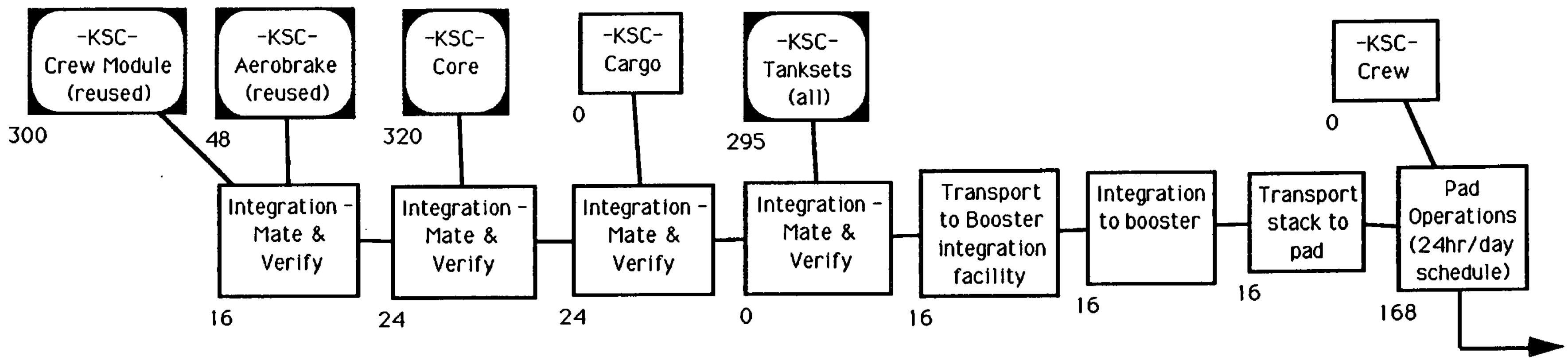
REFURBISHMENT OF LUNAR VEHICLE FOR NEXT MISSION



- **Vehicle assembly at base**
- **Vehicle element transport from KSC to base**
- **Vehicle element processing at KSC**

Figure 2-1.1.2.4-3. Example Operations Flow - SB2-1.5S (Sheet 3 of 3)

PROCESSING A GROUND BASED VEHICLE AT KSC



- **Ground based vehicles have much simpler flows in preparation for a mission compared to space based**
  - **Eliminates multiple booster integrations**
- **Space based vehicles allow the Lunar Mission "Go/No Go" decision to be made before leaving base**

Observation will be included in further considerations on base location

Figure 2-1.1.2.4-4. Example Operations Flow - GB2-1.5S (Sheet 1 of 2)

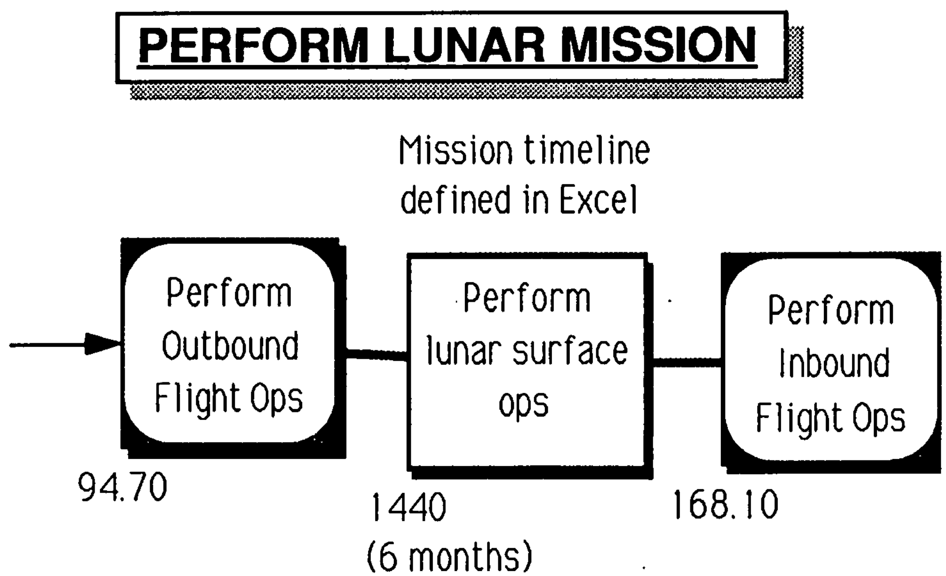


Figure 2-1.1.2.4-4. Example Operations Flow - GB2-1.5S (Sheet 2 of 2)

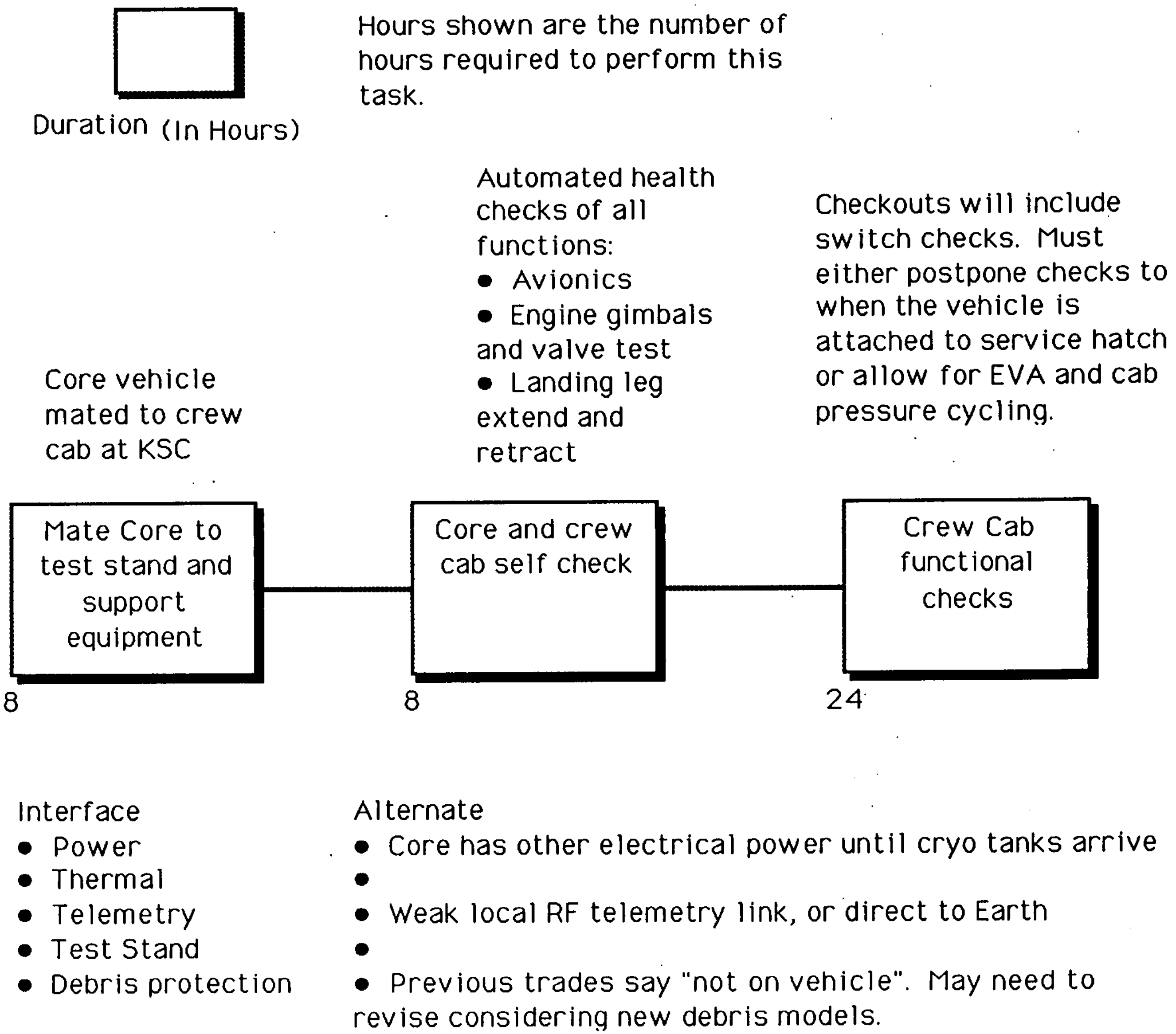
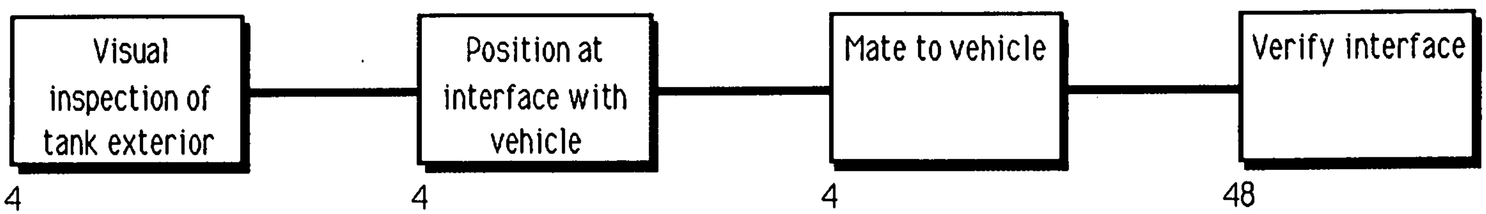


Figure 2-1.1.2.4-5. Example Operations Subflow - Stage and Cab (Sheet 1 of 2)

Snap in tanks

May have a requirement for "continuious monitoring" of pressure. How is this provided during ETO Transport phase?



INTERFACE DESCRIPTION:
- Mechanical attachment is by commanded latch.
- Manifold connection is by self sealing commanded latch.
- Zero-force, automatic mate, electrical connectors.
- SSF robotics capable of 70t objects.

Rationale: 1. Need simple, cost effective interface.
2. Need reusable, jettisonable interface (drop tanks and tanks that are remated in LLO)

- Part of the interface may not be verified at this time. May be required to keep fluids in tanks (isolation valves). Need some method of verifying plumbing connection.
  - Included here is a 48 hour (around the clock) pressure decay test on manifold connections.

Figure 2-1.1.2.4-5. Example Operations Subflow - Stage and Cab (Sheet 2 of 2)

2-1.1.3 System Architecture Trade Study Evaluation

In general, the scenarios were evaluated against each of the four evaluation criteria and then normalized to a 1 to 5 scale with 1 being the best score and 5 being the worst score. The total score was then developed as a summation of the score for each criterion times the weighting of each criterion. This total score was then normalized, or respread, to a 1 to 5 scale again with 1 being the best and 5 being the worst. The evaluation criteria followed by the weighting for each criterion as agreed on with MSFC were as follows cost (50%), margins and risk (30%), other mission capture (15%), and benefits to Mars (5%). Details of the evaluation methods for each of the criteria will be discussed in the following paragraphs.

Forty-three scenarios were finally evaluated. These scenarios were selected based on trends seen from the initial characterization of the 94 combinations and the initial cost and margins and risk evaluations of 29 scenarios. The scenarios selected for full evaluation were evaluated against one another in total and were additionally chosen to allow coverage of the individual architecture trades. For example, the number of stages was evaluated for multiple basing options and for the use of the lunar orbit rendezvous and the direct-to-the-lunar-surface lunar approach trajectories. Using this approach, a single scenario could be used in combinations with different other scenarios to support evaluations of several architecture trades. Figure 2-1.1.3-1 shows how the 43 scenarios were used in various combinations to fully exercise the trades with a minimum expenditure of study resources. Both the overall scenario scores along with the scenario comparisons for each trade were used in the downselection process.

Cost Assessment. The cost score was based on a combination of 70% design, development, test, and evaluation (DDT&E) costs and 30% LCC. This approach was based on the belief that the DDT&E costs, being the driver behind the level of funding required to obtain a new program start, should be strongly emphasized. All scenarios met the basic mission requirements, so an affordable funding profile at the beginning of the program, which would facilitate a program start, was seen as a valid discriminator.

43 Scenarios fully evaluated and used in combinations	
SB1-1.5S	GO2-2.5S
SB1-1.5SP	GO2-2.5H
SB1-2.5S	GO2-1.5S
	GO2-1.5H
SB2-1.5S	
SB2-1.5H	SG1-1.5S
SB2-2H	
SB2-2.5H	SG2-1.5S
SB2-2.5D	SG2-1.5H
SB2-2.5S	SG2-2H
SB2-2.5HP	SG2-2.5H
SB2-3H	SG2-2.5HP
SB2-3.5H	SG2-3H
SB2-4H	
	GB1-1.5S
GO1-1.5S	GB1-2.5S
GO1-1.5H	
GO1-2H	GB2-1.5H
GO1-2S	GB2-1.5S
GO1-2.5H	GB2-2H
GO1-2.5S	GB2-2.5H
GO1-3H	GB2-2.5D
GO1-3S	GB2-2.5S
	GB2-3H
	GB2-3.5H
	GB2-4H

BASING

SB2-2.5H	SB2-1.5S
SG2-2.5H	SG2-1.5S
GO2-2.5H	GO2-1.5S
GB2-2.5H	GB2-1.5S
SB1-1.5S	SB1-2.5S
SG1-1.5S	GO1-2.5S
GO1-1.5S	GB1-2.5S
GB1-1.5S	

SB2-1.5H
SG2-1.5H
GO2-1.5H
GB2-1.5H

STAGE NUMBER

Ground Based (LLO)	*Space Based (LLO)*
SB2-1.5H	GB2-1.5H
SB2-2H	GB2-2H
SB2-2.5H	GB2-2.5H
SB2-3H	GB2-3H
SB2-3.5H	GB2-3.5H
SB2-4H	GB2-4H

Comb. Space/Ground Based (LLO)

SG2-1.5H
SG2-2H
SG2-2.5H
SG2-3H

Ground Based LEO Rendezvous (LS Direct)

GO1-1.5H
GO1-2H
GO1-2.5H
GO1-3H

Figure 2-1.1.3-1. Scenarios for Evaluation (Sheet 1 of 2)

LLO vs LS DIRECT

SB1-1.5S	SB1-2.5S
SB2-1.5S	SB2-2.5S
SB2-1.5H	SB2-2.5H
GB1-1.5S	GB1-2.5S
GB2-1.5S	GB2-2.5S
GB2-1.5H	GB2-2.5H
GO1-1.5S	GO1-2.5S
GO2-1.5S	GO2-2.5S
GO2-1.5H	GO2-2.5H
SG1-1.5S	
SG2-1.5S	
SG2-1.5H	

CREW MODULE APPROACH

SB2-2.5S	GB2-2.5S
SB2-2.5H	GB2-2.5H
SB2-2.5D	GB2-2.5D
SB2-1.5S	GB2-1.5S
SB2-1.5H	GB2-1.5H
SG2-1.5S	GO2-2.5S
SG2-1.5H	GO2-2.5H
	GO2-1.5S
	GO2-1.5H

SPACE BASED ONLY

AEROBRAKE vs ALL-PROPULSIVE

SB2-2.5H	SG2-2.5H
SB2-2.5HP	SG2-2.5HP
SB2-1.5H	SG2-1.5S
SB2-1.5HP	SG2-1.5SP
SB2-1.5S	SG2-1/5H
SB2-1.5SP	SG2-1.5HP
SB1-1.5S	
SB1-1.5SP	

DROP TANKS vs TANKERS (Tank Reuse)

SB2-2H	SG2-2H
SB2-2.5H	SG2-2.5H

Figure 2-1.1.3-1. Scenarios for Evaluation (Sheet 2 of 2)

Additionally, costs were attributed to some scenarios based on the assumption that the STV program would be responsible for a portion of the development of other program hardware. All GB scenarios would require very large booster capability, on the order of a 260 metric ton LEO payload delivery capacity. Based on the assumption that STV would be the first and primary user (until the Mars program) of the system, GB scenario cost scores included a $7 billion cost for facilitization and other system impacts required for the booster. For the SB and SG scenarios that would use a LEO node, a $4.5 billion cost was included in the cost scoring. This was taken from the General Dynamics Space Transportation Infrastructure Study (STIS) for estimated LEO node costs to modify the SSF to accommodate SEI missions. The GD STIS estimate was broken down into top-level elements and examined and accepted as a reasonable estimate.

Figure 2-1.1.3-2 shows the process used to develop the LCC for each scenario. In summary, all applicable costs were developed for each flight element and the LCC model used these costs in conjunction with boost costs and the Option 5 manifest to develop the LCC. The LCCs were developed with both a high-boost cost of $6,000/kg ($2,721/lb) and a low-boost cost of $1,000/kg ($454/lb). For the evaluation process, the LCCs based on low-boost costs were used with the justification that a low-cost ETO system would be necessary for the masses required to be delivered to ETO in support of the SEI program. Additionally, the low-boost costs were in the range of the Advanced Launch System (Alunar surface) goals for ETO delivery costs. The LCCs based on high-boost costs were used to determine sensitivities of the selections to boost cost. The cost development process is described in more detail in the following paragraphs.

For this phase of the study, the space-based, LLO node, 1.5-stage, single crew module scenario was used as a reference case. The Boeing-developed parametric cost model (PCM), vehicle processing flows, and information from a variety of other programs were used to define costs for each unique flight element in terms of a costing variable. Costs were developed for all the cost elements, such as development, production, processing, and refurbishment, relevant to that flight element type.

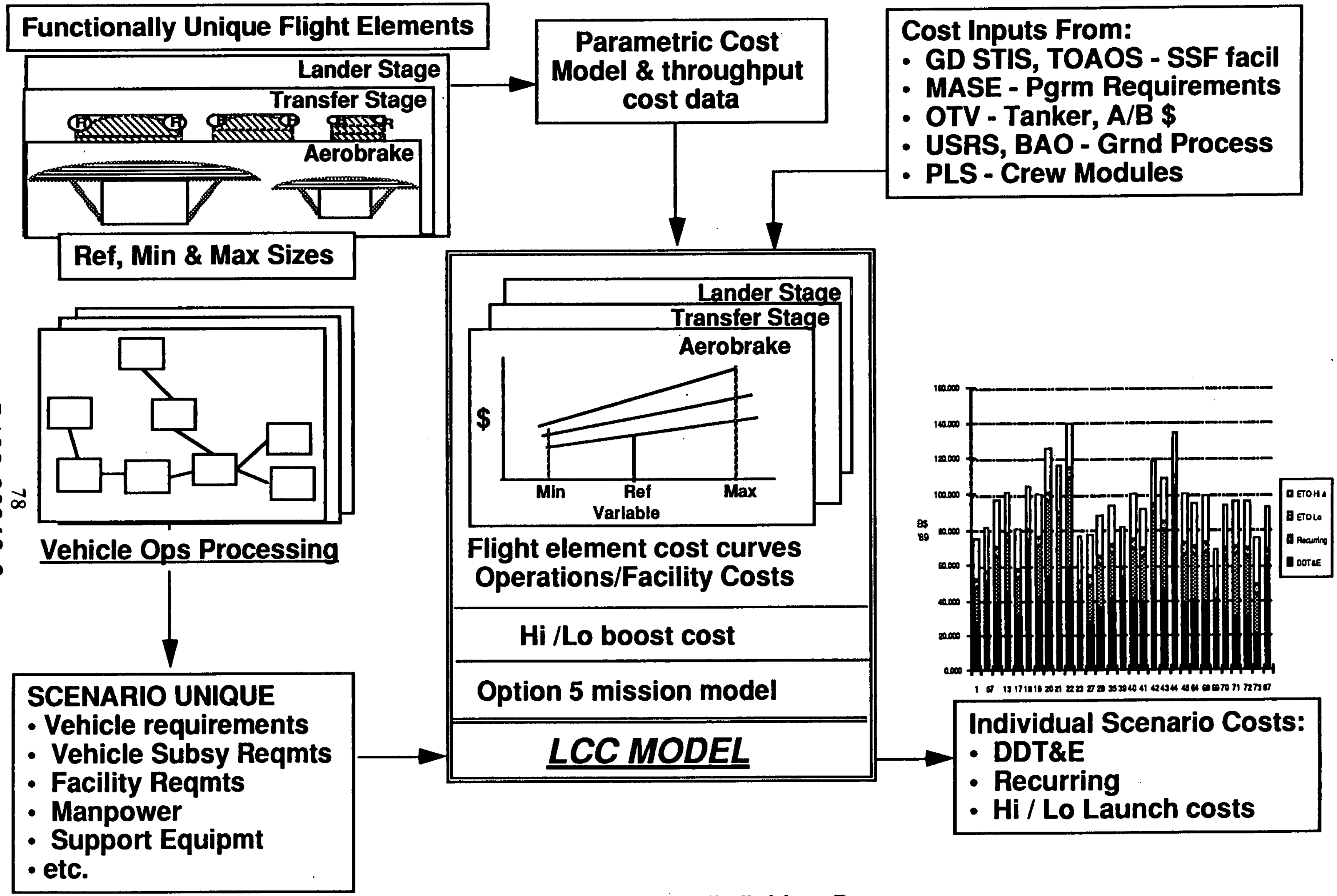


Figure 2-1.1.3-2. Cost Definition Process

Cost elements (e.g., development, refurbishment, and production) for each flight element (e.g., the aerobrake) were developed for the reference case element and for the largest and smallest elements (e.g., reference SB2-1.5S aerobrake and largest and smallest aerobrakes across all scenarios). These three "point design" costs were then used to generate cost look-up curves based on the costing variable in the LCC model for each cost element for each flight element. For example, inert mass was used as the costing variable for the aerobrake and cost look-up curves were generated for development, space refurbishment, build, ground processing, assembly, and so forth based on inert mass. Relevant costs for a scenario unique aerobrake could then be generated using the inert mass of that aerobrake and the curves.

The LCC model then used these cost elements, boost costs per flight element per mission, and the number of each kind of flight (steady state, replacement, and expendable cargo) from the Option 5 mission model along with the non-recurring costs and scenario to determine the overall LCC for each scenario.

Margins and Risk Assessment. The STV system and each of the subsystems will be designed with margins for all contingencies. In addition, risks for each mission operation and each mission phase will be mitigated as much as possible using modern engineering techniques. However, some system configurations will inherently have margins and some system configurations will inherently mitigate risks simply because the architecture avoids particular situations during the mission profile. The margins and risk evaluation attempted to identify and quantify the risks and margins that are discriminators between the scenarios.

The breakdown in weighting between risk and margins and the respective subcategories is shown in Figure 2-1.1.3-3. The risk area is broken into equal weighting between technical and programmatics risk. Technical risks deal with the risk during the operational phase and include such things as mission success, performance and operation, and safety and reliability. In general, the programmatic risk deals with the anticipated risk associated with the FSD program phase (i.e., cost and schedule). The technical risk category is further broken into 10 risk subcategories as shown in Figure 2-1.1.3-4, which are weighted as to their respective importance. Each system concept was given

Evaluation criteria	
Cost	50%
Risk & margin	30%
Evolutionary mission	15%
Mars mission	5%

Risk categories	Risk weight = 20%
Technical	50%
Programmatic	50%

Margin categories	Margin weight = 10%
Mission capture	20%
Payload growth	35%
Operational flexibility	35%
Safety	5%
Repairability	5%

Figure 2-1.1.3-3. Risk and Margin Category Weighting

Technical	Weight = .50
• Space environment exposure time	.04
• In space assemble	.12
• Number of rendezvous	.02
• LEO node support	.02
• LLO node support	.02
• Micro g's propellant transfer	.04

Technical (continued)	
• Abort penalties	.12
• Dual ECLSS system available	.08
• Reuseable motor/CC	.02
• Reuseable aerobrake	.02

Programmatic	Weight = .50
• Cost	
• Schedule	.20
• Management	
• Development	.30

Figure 2-1.1.3-4. Risk Category Weights

relative grades of either 1, 2, or 3 (1 for low risk, 2 for medium risk, and 3 for high risk) for each of these categories with low risk being best. Figure 2-1.1.3-5 contains the detailed definitions and respective scoring approach for all of the risk categories. The risks evaluated here exclude design for risk mitigation.

The five margin categories (mission growth, payload growth, operational flexibility, safety, and repairability) and the scoring rationale are shown in Figure 2-1.1.3-6. The margins evaluated here exclude design margins.

Mission Capture Assessment. Evolutionary mission capture was one of the evaluation criteria for STV concept selection with a weighting of 15% of the total evaluation criteria. The purpose of this analysis was to determine how well the STV concepts designed for the lunar missions could capture other NASA and DoD missions identified as design reference missions (DRM). Those missions included:

1. 16,000-kg planetary.
2. 10,000-kg unmanned GEO delivery.
3. 6,800-kg molniya delivery.
4. 4,000-kg manned GEO servicing.
5. 4,500 kg polar platform servicing.
6. 25,000-kg nuclear/debris disposal.
7. 500-kg manned capsule recovery.

A general groundrule used for this analysis was that only "smart" stages based at the SSF or the ground could be used as the primary stage for these other missions.

The concepts were scored both by stage efficiency, that is, how efficient the lunar-sized stage can perform the other missions (required propellant mass and total start mass, excluding payload), and by Earth-to-orbit launched mass. These values were averaged over the mission model by the percentage of each mission included and then scored 1 to 5 (1 = best and 5 = worst) and weighted 80% mass fraction (i.e., stage efficiency) and 20% ETO mass. This weighting accentuates the stage efficiency in performing other missions. Because the NASA-only mission model and the combined NASA/DoD mission model differ

Technical Risk - Part I	Score	Sample discriminators
Time exposure to environment between refurb/support	=1 during mission flight only =2 for 6 month LLO =3 for multiple 6 month periods	LLO nodes stored multiple 6 month periods versus direct ascent and descent
In-space assemble	=1 GB systems =2 5 or less mating surfaces =3 6 or more mating surfaces	GB versus GO versus SB
Number of rendezvous	=1 no mating =2 1 mating =3 2 matings or more	LLO & SSF rendezvous require in space mating. (mating of launcher & lunar surface required by all concepts)
Node support in LEO required	=1 independent of LEO support =2 on orbit assemble =3 LEO support mandatory	Mission critical support received for LEO node versus GB
Nodes support in LLO required	=1 independent of LLO support =3 LLO support mandatory	Mission critical support received for LLO node versus direct LS
Micro g's propellant transfer	=1 none required =3 required to accomplish mission	Systems with no transfer are judged to have less risk

Low = 1 (low risk is better) Medium = 2 High = 3

Figure 2-1.1.3-5. Scoring Approach - Risk Categories (Sheet 1 of 4)

Technical Risk - Part II	Score	Sample discriminators
Abort penalty	=1 GB, SB GO w/free return =2 SB w/3 or fewer stages =3 SB w/more than 3 stages	Each mission phase has abort option–this category measures the penalty (see next chart)
Dual ECLSS system available	=1 Hybrid CC =3 Dual or single CC	Hybrid cab versus dual or single cab configuration
Reusable motor and CC	=1 multiple reuse =3 no reuses	Avionics and shuttle engine reuse has shown history or reliability confidence
Aerobrake reuse	=1 no reuse =3 multiple reuses	No multi use aerobrake currently available without GB checkout and refurbishment

Low = 1 (low risk is better) Medium = 2 High = 3

Figure 2-1.1.3-5. Scoring Approach - Risk Categories (Sheet 2 of 4)

Abort Scoring

Mission phase	Description of penalties and sample variation	Points
Pre TLI	No abort penalty, see margins–operational penalty	
Translunar flight	GO and GB have ballistic cab and allow free return to earth surface Space based cab have plane change propellant penalty	1
Post LOI (If applicable)	Return to SSF implies multi-day wait in LLO or large delta-V penalty versus ballistic cab required LLO waits of less than an hour	2
Landing	Ascent stage in multi-stage system provide protection against landing debris and engine fratricide losses. Light weight CC enhance abort capability	3
Surface stay	SSF node forces large delta-V penalty	4
Ascent	Return to lunar surface is a problem for 4 stage concept	5
LLO rendezvous (If applicable)	Positive margin for ECLSS system stored in LLO (hybrid and dual CC)	6
Post TEI		
Aerobrake/reentry (If applicable)	Safety issue no abort capability–this maneuver not required for all propulsive.	7
LEO circularization (If applicable)	EOI–Aerobrake systems could use the RCS of MPS for this maneuver. Aerobrake slightly better than all propulsive	8
SSF rendezvous (If applicable)	Additional maneuvers required for SSF rendezvous.	9

Figure 2-1.1.3-5. Scoring Approach - Risk Categories (Sheet 3 of 4)

Programmatic Risk Breakdown

Development category	Score	
Return	=1	all propulsive to LEO
	=2	GB reentry
	=3	aerobrake to LEO
Assemble and checkout	=1	GB
	=3	SB
Micro g's propellant transfer	=1	none
	=2	In LEO
	=3	In LEO and LLO
Number of FSEDs	=1	1 or 2
	=2	3 to 5
	=3	5 or more
Software for checkout	=1	GB
	=3	SB
O&M methods and procedures	=1	GB
	=3	SB

Simple arithmetic mean of sum of scores for each system

Figure 2-1.1.3-5. Scoring Approach - Risk Categories (Sheet 4 of 4)

Margin category	Score	Sample discriminators
Mission capture	=1 GB systems =3 SB or GO systems	GB limited by launch system capacity
Payload growth	=1 34 MT or less =2 35 to 44 MT =3 45 MT or more	Estimates of payload capability to lunar surface resulted in these three catecories.
Operational flexibility	=1 otherwise =2 If daily but LLO rendezvous =3 if launch &return opportunities are daily	GB launch cancellation results in 24 hour delay SB launch cancellation results in 6 to 11 days delay
Safety	=1 19 or more points =2 12 to 18 points =3 11 points or less	Multiply the number of each operation times the complexity of the operation. Operation (complexity): Rendezvous (2) aero-maneuver (10), Ballistic return (5), EVA (4), fuel transfer (3), proximity (1)
Repairability	=1 =2 =3	Rated as 2 for this screening

Low = 1 Medium = 2 High = 3 (high margins are better)

Figure 2-1.1.3-6. Scoring Approach - Margins Categories

as to the types of missions that were included, the analysis was done for each mission model and they were given equal weight in this analysis. Figure 2-1.1.3-7 provides an overview of the categories and category weightings used for this analysis.

Figure 2-1.1.3-8 tabulates the mission capture calculations for a sample vehicle concept, namely the SB2-1.5S (space-based with LLO rendezvous, 1.5-stage, single crew module). The stage efficiencies for the various missions vary from 0.458 (least efficient) for the manned capsule delivery to 0.851 (most efficient) for the polar platform servicing mission. ETO mass varies from 24,686 to 150,803 kg for the same missions.

The values for stage efficiency and ETO mass were averaged over the mission models, scored on a scale from 1 to 5 (1 = best), and weighted 80% and 20%, respectively. These scores were then weighted 50/50 for the NASA and NASA/DoD mission models for an overall score of 3.211.

Benefits to Mars Assessment. The Mars mission benefit was one of the evaluation criteria for STV concept selection with a 5% weighting of the total evaluation criteria. The purpose of this analysis was to determine how much the STV concepts, designed for the lunar missions, can benefit the Mars missions and vehicle designs as they are projected at the current time.

Mars vehicle designs include a transfer vehicle (MTV) and an excursion vehicle (MEV). MTV options include cryogenic vehicles, nuclear energy propulsion (NEP) vehicles, solar energy propulsion (SEP) vehicles, and nuclear thermal rocket (NTR) vehicles. For this analysis, it was assumed that the MEV is cryogenic and has an aerobrake, no matter what the MTV type. Because the cryogenic MTV would benefit most from the lunar missions, it was chosen as the baseline for this analysis. To determine the overall benefit of each of the lunar vehicle concepts, specific benefits were weighted independently and scored and then combined with equal weighting for the MTV and MEV.

Types of Mars mission benefits were broken into subsystem- and system-level benefits (e.g., structures, aerobrake, and propulsion) and further into specific areas of benefit (e.g., landing gear, mate and demate umbilicals, and aerobrake

Overall Weighting	Evolutionary Mission Capture	15%
Mission Model Weighting	NASA - Only Mission Capture = 50%	
	NASA / DoD Mission Capture = 50%	
Scoring Criteria	Average Stage Efficiency	80%
	Average ETO Mass	20%

Mission Model Split

Mission	NASA-only	NASA / DoD
P1 Planetary Mission	23%	4%
G1 GEO Unmanned Delivery	31%	51%
D1 Molniya Delivery	0%	37%
G2 Manned GEO Servicing	12%	2%
S1 Polar Platform Servicing	15%	2.5%
N1 Nuclear / Debris Disposal	4%	1%
C1 Manned Capsule Recovery	15%	2.5%

Figure 2-1.1.3-7. Mission Capture Assessment

Absolute Values:

Sample Vehicle: SB2 - 1.5S

Mission	Stage Efficiency	ETO Mass - kg	NASA-only	NASA / DoD
P1 Planetary Mission	.762	103105	23%	4%
G1 GEO Unmanned Delivery	.772	68836	31%	51%
D1 Molniya Delivery	.771	65105	0%	37%
G2 Manned GEO Servicing	.728	96399	12%	2%
S1 Polar Platform Servicing	.851	150803	15%	2.5%
N1 Nuclear / Debris Disposal	.750	107317	4%	1%
C1 Manned Capsule Recovery	.458	24686	15%	2.5%

Combined Scores (1=Best, 5=Worst)

Scoring Criteria		NASA-only (50%)	NASA / DoD (50%)
Average Stage Efficiency	80%	3.325	2.558
Average ETO Mass	20%	3.882	3.346

Total Score : 3.211

Ranking: 33rd on NASA-only missions, 27th on NASA/DoD missions, 33rd overall out of 42

Figure 2-1.1.3-8. Mission Capture Scoring Example

on-orbit assembly) and then weighted independently for the MTV and MEV. These were then graded as to the level of benefit received from the lunar mission technologies (1 = technology benefit and 2= hardware or operations benefit). Figure 2-1.1.3-9 shows the areas of benefit and weighting for each of these areas.

The Mars vehicle weighting for each system or subsystem item was multiplied by the lunar vehicle benefit and summed to achieve a total score for each lunar vehicle concept. The scores for the MTV and MEV were then weighted equally to yield the overall Mars benefit score for each lunar vehicle option.

Figure 2-1.1.3-10 tabulates the Mars benefit calculations for a sample vehicle concept, namely the SB2-1.5S (space-based with LLO rendezvous, 1.5-stage, single crew module). The total combined points for each of the Mars vehicle concepts and for each of the system and subsystem areas are shown. Areas such as structures, avionics, and robotics show high benefits to all Mars vehicle types. The total weighted MTV (cryogenic) and MEV benefit scores for all lunar concepts are then ranked on a 1 to 5 scale (1 = best). The total score for this vehicle was a 1.334, indicating very good benefits for the Mars missions.

2-1.1.4 System Architecture Trade Study Results

The System Architecture Trade study resulted in a downselect to three architecture options for further definition. All of the scenarios were 1.5-stage vehicles using a single crew module and all used the lunar orbit direct trajectory approach to lunar landing. The main difference in the three scenarios was in the basing. One scenario was ground based with single launch, one was ground based with on-orbit assembly through rendezvous and docking, and the final scenario was a space-based architecture. For the space-based case, droptanks were used instead of propellant tankers, and an aerobrake was used for return to the SSF.

Figure 2-1.1.4-1 contains the scores for the scenarios, both by individual evaluation criterion and by total score. Note that, when lunar orbit rendezvous is deleted (due to the safety and abort considerations), the three downselected scenarios are the three with the best (i.e., lowest) total scores. Both the overall

Overall Weighting

Benefit to Mars Missions	5%

Mars Vehicle Type Weighting

	MTV (50%)				MEV (50%)
	CRYO	NEP	SEP	NTR	CRYO
Structures and Mechs	8%	17%	19%	8%	8%
Aerobrake	15%	0%	0%	8%	16%
Propulsion	16%	16%	16%	12%	16%
Propellant Management	12%	0%	0%	12%	12%
Power	4%	30%	28%	13%	4%
Avionics	8%	8%	8%	8%	8%
Habitation / Crew Interface	8%	8%	8%	8%	8%
Robotics	10%	10%	10%	10%	9%
Integration and Test Ops	6%	6%	6%	6%	7%
Transportation Infrastructure	13%	5%	5%	15%	12%

Not used for scoring (shaded: NEP, SEP, NTR)

Lunar Vehicle Benefit

Technology-only Benefit	1
Hardware or Operations Benefit	2

Mars Vehicle Weighting X Lunar Vehicle Benefit = Mars Benefit Score

Figure 2-1.1.3-9. Benefits to Mars Assessment

Sample Vehicle: SB2 - 1.5S

Absolute Score by Subsystem (high is good)	MTV (50%)				MEV (50%)
	CRYO	NEP	SEP	NTR	CRYO
Structures and Mechs	14.3	16.4	16.4	14.3	14.8
Aerobrake	18.5	0.0	0.0	9.8	16.5
Propulsion	22.0	0.0	0.0	1.5	22.0
Propellant Management	28.0	0.0	0.0	28.0	28.0
Power	4.0	4.0	8.0	2.0	4.0
Avionics	17.5	17.5	17.5	17.5	17.5
Habitation / Crew Interface	9.1	9.1	9.1	9.1	11.1
Robotics	20.0	20.0	20.0	20.0	18.0
Integration and Test Ops	9.0	9.0	9.0	9.0	9.0
Transportation Infrastructure	16.0	5.0	5.0	19.0	14.0
Total:	158.4	81.0	85.0	130.2	154.9

Not used for scoring

Absolute score (High is good)	156.7	118.0	120.0	142.5
Ranking (out of 42 lunar veh types)	14 th	14 th	14 th	14 th
Overall Spread Score (on 1 to 5 Scale, 1= best):	**1.334**			

Figure 2-1.1.3-10. Benefits to Mars Scoring Example

Mission Scenario	Cost	Margins & Risk	Mission Capture	Benefits to Mars	TOTAL SCORES
GO2-1.5S	1.11	2.15	3.03	3.52	1.00
GB1-1.5S	1.68	1.00	3.30	5.00	1.08
GO2-1.5H	1.32	2.07	2.95	3.44	1.09
GO1-1.5S	1.00	2.32	3.28	4.16	1.09
SB1-1.5SP	1.46	3.36	1.35	2.76	1.35
GB2-1.5H	2.00	1.60	2.96	4.28	1.44
GB2-1.5S	1.80	1.99	3.14	4.36	1.51
SB2-1.5HP	1.70	4.05	1.31	2.04	1.77
SB2-1.5SP	1.48	4.62	1.04	2.11	1.80
SB1-1.5S	1.43	3.79	3.07	1.98	1.83
SG2-1.5HP	1.90	3.59	2.04	1.97	1.86
SG2-1.5SP	1.71	4.03	1.92	2.05	1.90
GO2-2.5H	2.63	1.76	3.27	3.25	1.96
GO1-2S	2.22	2.49	3.28	4.16	2.05
GO2-2.5S	2.25	2.58	3.33	3.33	2.06
SG2-1.5S	1.84	3.81	3.57	1.27	2.20
GB1-2.5S	2.96	1.24	4.21	5.00	2.30
GO1-2.5S	2.28	2.60	4.20	4.16	2.33
GB2-2.5S	2.93	2.03	3.47	4.17	2.39
SG1-1.5S	2.00	3.50	4.46	1.91	2.41
SB2-2.5HP	2.85	3.98	1.00	1.85	2.48
SG2-2.5HP	3.01	3.66	1.23	1.78	2.50
GB2-2.5H	3.39	1.64	3.28	4.09	2.51
SB2-1.5S	1.61	4.87	3.80	1.33	2.54
SG2-1.5H	2.01	4.35	3.66	1.19	2.57
GB2-2.5D	3.55	2.00	3.23	4.09	2.78
SB1-2.5S	2.73	3.87	3.00	1.98	2.79
GB2-3H	3.87	1.68	3.33	4.09	2.88
GB2-2H	3.43	1.60	5.00	4.09	2.90
SB2-1.5H	1.95	4.72	4.86	1.26	2.95
SB2-2.5S	2.53	4.94	2.89	1.14	3.02
GB2-3.5H	4.12	1.67	3.43	4.09	3.09
SG2-2.5H	3.06	4.60	2.63	1.00	3.19
SB2-2.5H	2.88	4.37	3.80	1.07	3.22
SG2-2H	3.29	4.00	3.32	1.00	3.25
GO1-3S	3.41	2.58	4.89	4.16	3.28
GB2-4H	4.61	1.67	3.52	4.09	3.46
SB2-2.5D	3.04	4.86	3.75	1.07	3.53
SB2-2H	3.10	5.00	3.49	1.07	3.58
SG2-3H	4.35	4.10	2.81	1.00	3.94
SB2-3.5H	3.72	4.55	4.15	1.07	3.97
SB2-3H	4.16	4.60	4.21	1.07	4.32
SB2-4H	5.00	4.54	4.66	1.07	5.00

Figure 2-1.1.4-1. System Architecture Trade Scores

scenario scores and the scenario comparisons for each trade were used in the downselection process.

One of the findings in this trade was that better performance did not necessarily equate to lower costs. Better performing systems tend to have higher development and operations costs that outweigh the higher propellant delivery costs associated with lower performing systems. Figure 2-1.1.4-2 illustrates this finding with comparisons of LCC and performance (in terms of mass required in LLO for the lunar mission model). Note that the LCC numbers were top-level estimates developed for the System Architecture Trade Study and using top-level flight and operations elements only. Costs were subsequently refined for the downselected scenarios.

Two comparisons were made, one based on staging and one based on the lunar approach trajectory options of the lunar direct scenario and the LOR scenario. The staging comparison used space-based scenarios using LOR, one single stage with droptanks, and a single crew module (SB2-1.5S) and one dual stage with droptanks, and a dual crew module (SB2-2.5D) that was the 90-day study baseline. The comparison based on LOR versus LD used a space-based, single stage with droptanks and single crew module. One scenario used LOR (SB2-1.5S) and one scenario used LD (SB1-1.5S).

It becomes apparent that in the staging comparison, the two-stage vehicle (performance optimized in the 90-day study) has the better performance while also having the higher cost. This cost remains higher than the lower performance one-stage vehicle across a range of boost costs ($/kg to orbit) until the boost cost reaches approximately $53,000/kg.

In the LOR versus LD comparison, the use of LOR provides the best performance. Here the LD provides the best cost at the low end of the boost costs; however, there is a crossover point at a boost cost of approximately $2,240/kg where the LOR option becomes attractive (from a cost standpoint).

The primary reasons for better performance not necessarily equating to lowest cost is that a higher performing system tends to be more complicated with higher DDT&E and recurring costs as can be seen in Figure 2-1.1.4-3. In the

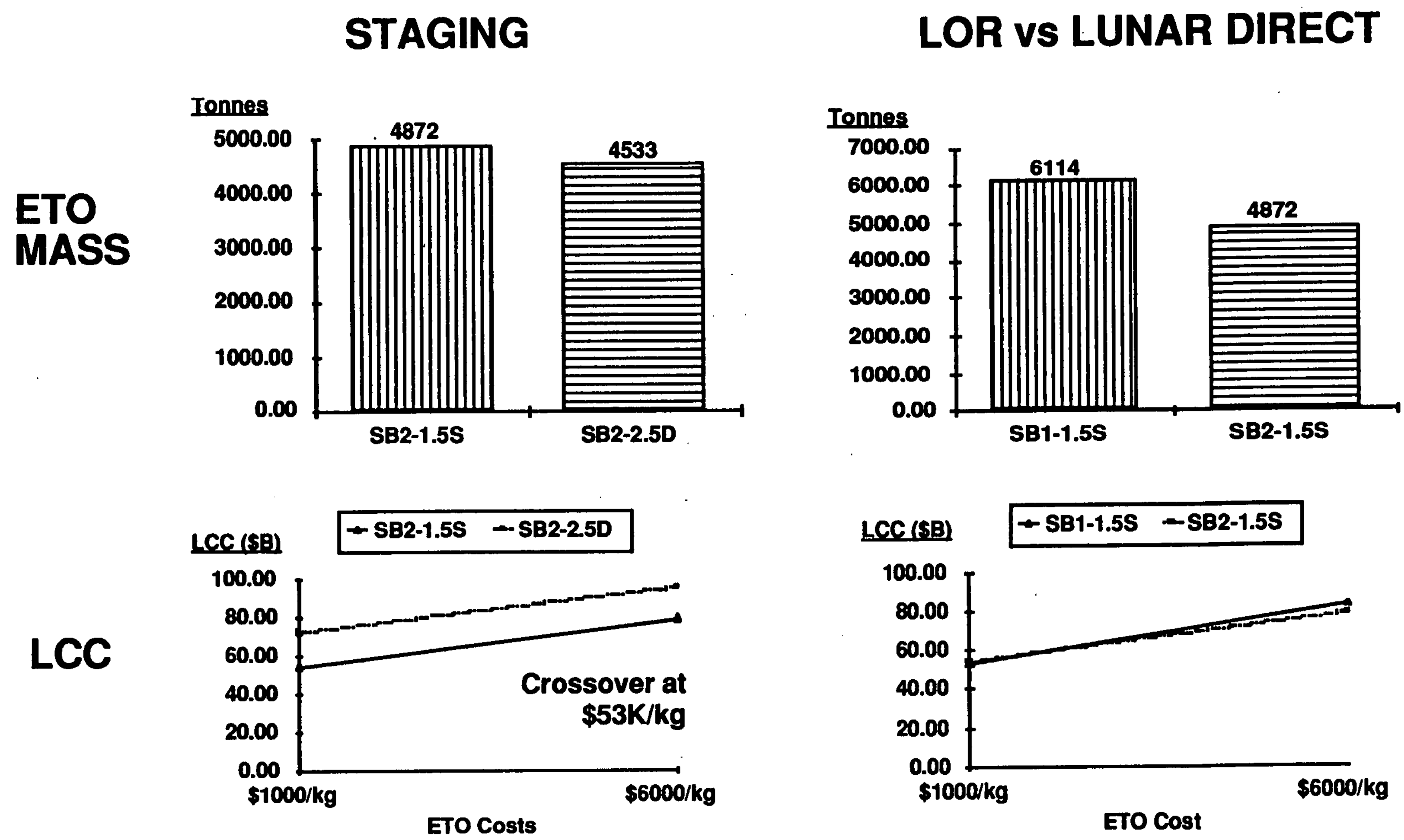
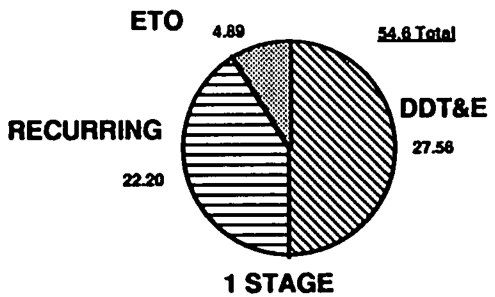
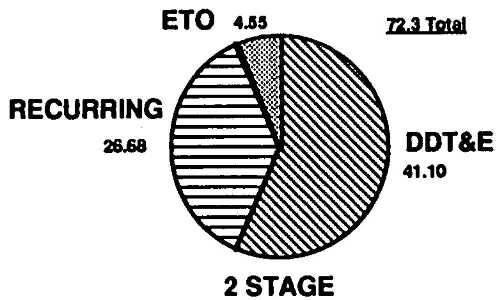


Figure 2-1.1.4-2. Cost Is Not Equal to Performance Examples

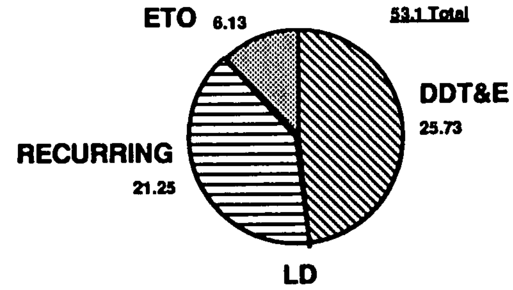
STAGING

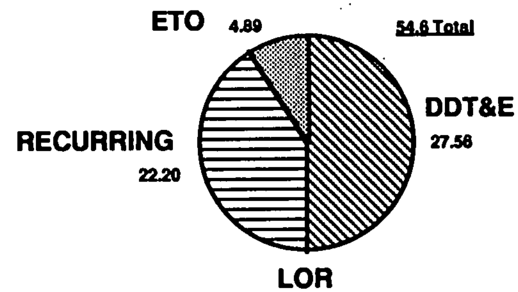




DELTA	
2 Stage - 1 Stage	
DDT&E	13.54
ETO	<0.34>
Rec.	4.48
TOTAL	17.68

LUNAR DIRECT (LD) vs LOR





DELTA	
LOR - LD	
DDT&E	1.83
ETO	<1.27>
Rec.	0.95
TOTAL	1.51

Figure 2-1.1.4-3. Cost Is Not Equal to Performance Cost Elements

staging example, the DDT&E and recurring (production and operations) costs are both higher for the two-stage vehicle than for the one-stage vehicle. Each of these costs alone are also a much higher percentage of the overall LCC than the boost cost and so are much more significant than the boost costs. The DDT&E and recurring costs are higher for the two-stage vehicle because it is a much more complicated vehicle with more elements required to be developed, assembled, and refurbished (where reused).

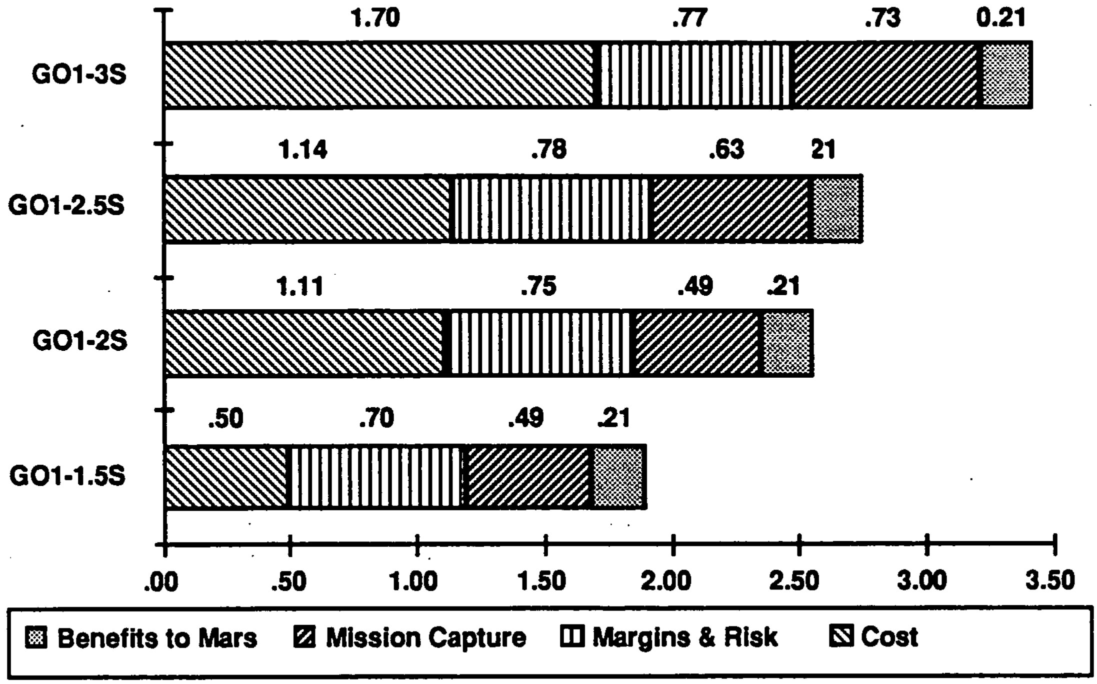
In the LOR versus LD comparison, the DDT&E and recurring costs are close enough that the LCC is sensitive to deltas in the boost costs. The primary reason why the use of LOR has a higher DDT&E cost is that the elements left in LLO during the mission (aerobrake and return propellant tanks) require stationkeeping equipment to maintain a stable low lunar orbit.

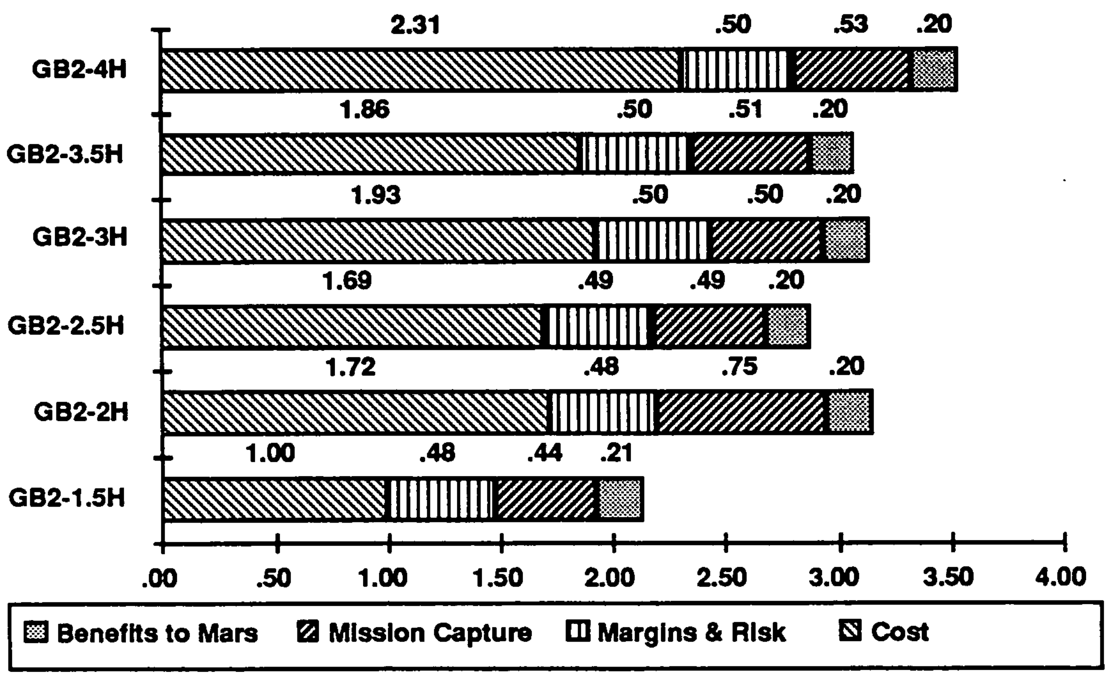
The following sections discuss the results for each of the architecture trade studies.

Number of Stages. The results of the scenarios compared for the number of stages trade strongly indicated that fewer stages were preferred, with the single-stage scenarios (with droptanks) being the clear winners. Although the single stages, in general, did not have the best performance, the reduction in operational complexity and development costs for the fewer stage vehicles outweighed the performance penalties.

The scoring for the scenarios used to compare the number of stages is shown in Figure 2-1.1.4-4. In this and the following figures showing the scores, the cost score is 50% of the total cost score based on the 1 to 5 scale, the margins and risk score is 30% of the total margins and risk score based on the 1 to 5 scale. Therefore, it is appropriate to compare cost scores between different scenarios but not to compare cost scores against mission capture scores. Note also that the total score obtained by adding the cost and margins and risk scores do not equal the total scores shown in Figure 2-1.1.4-1, where the total scores were respread to a 1 to 5 scale.

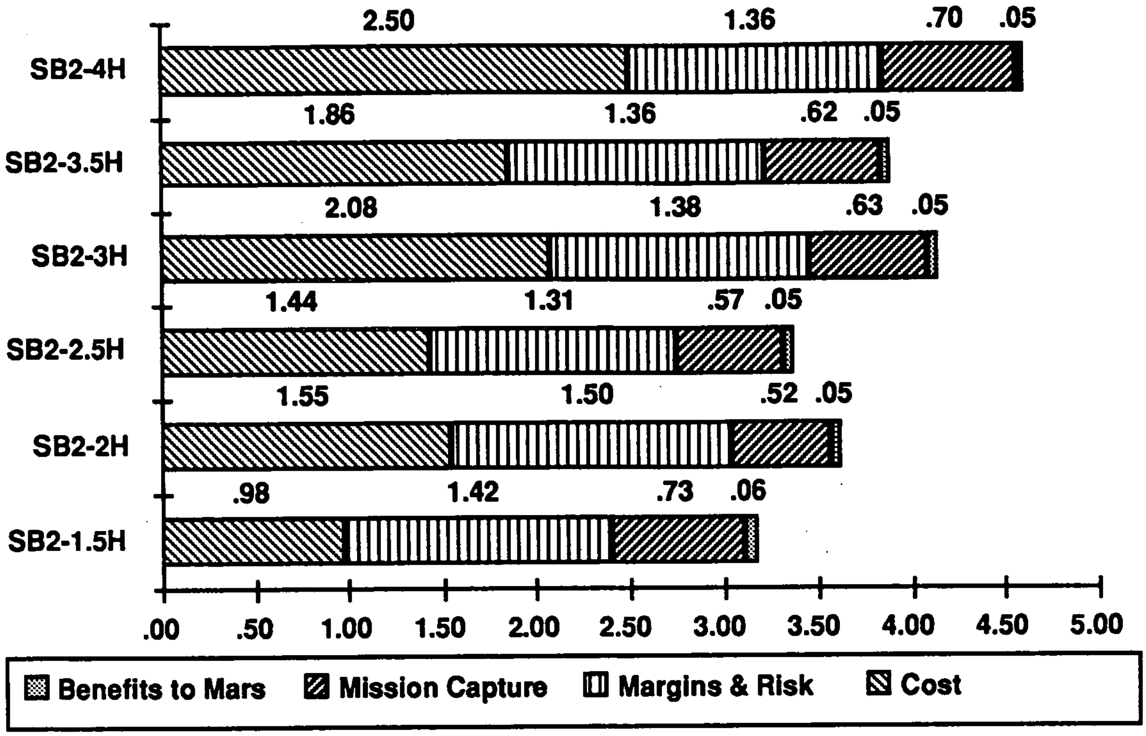
Crew Module Approach. The scoring for the scenarios used to compare crew module approaches, shown in Figure 2-1.1.4-5, indicates that the single

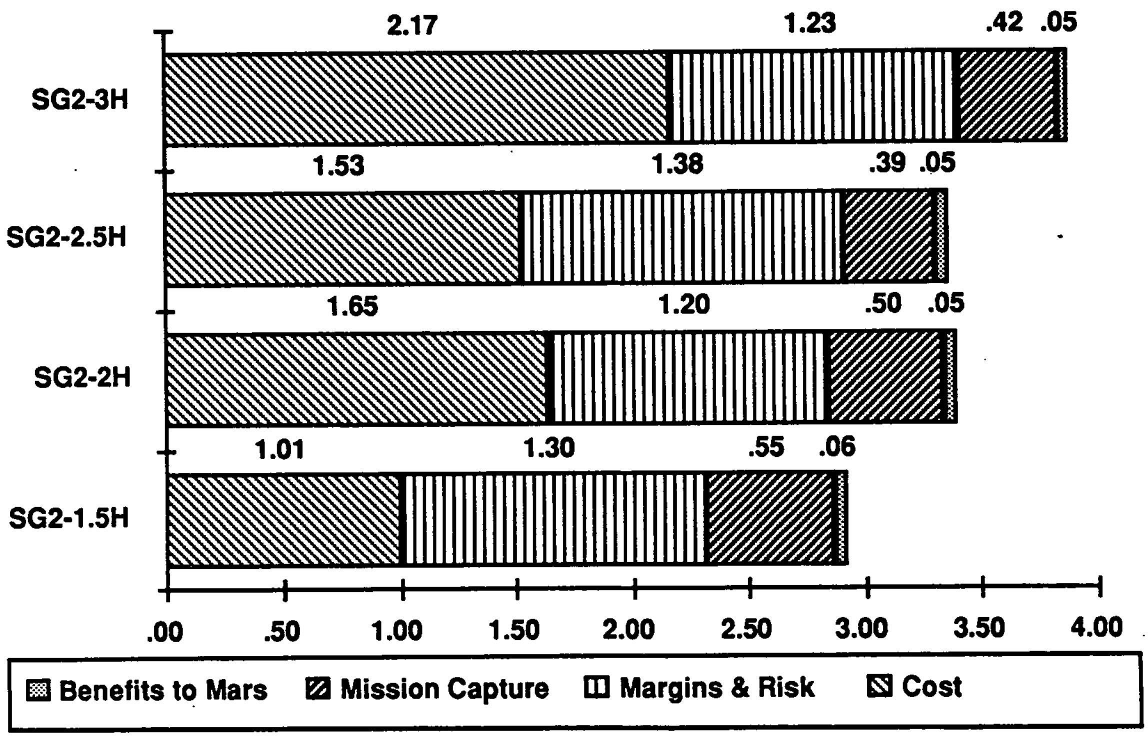




NOTE: Low scores are best

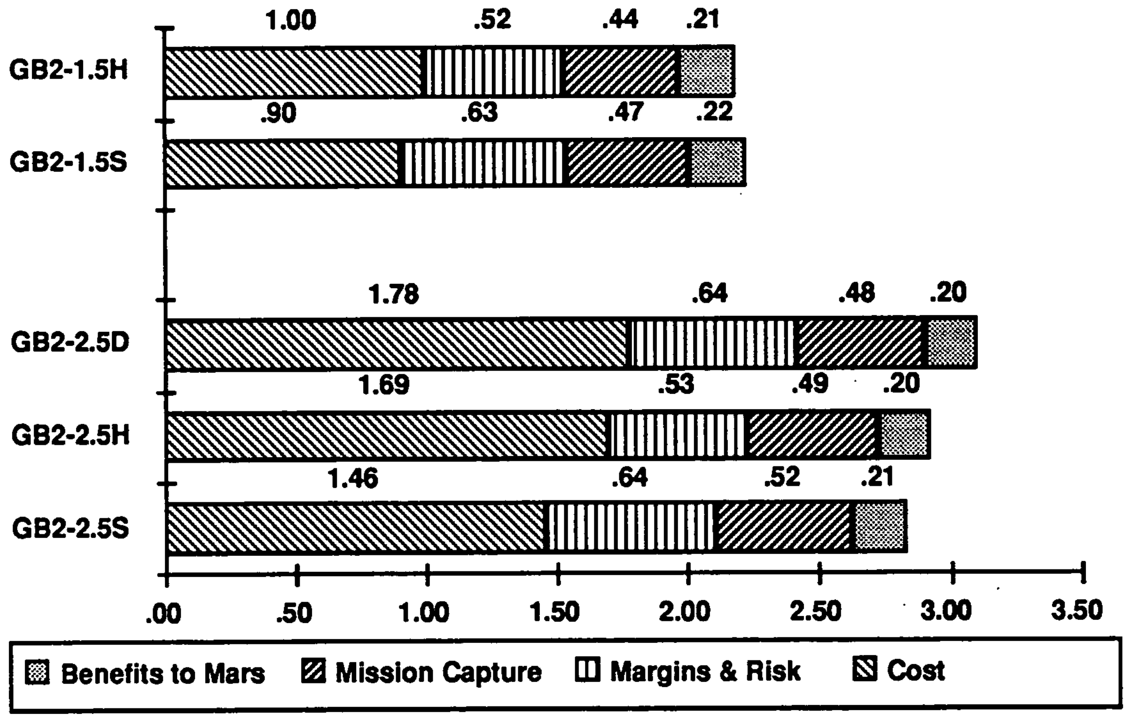
Figure 2-1.1.4-4. Number of Stages Trade Scores (Sheet 1 of 2)

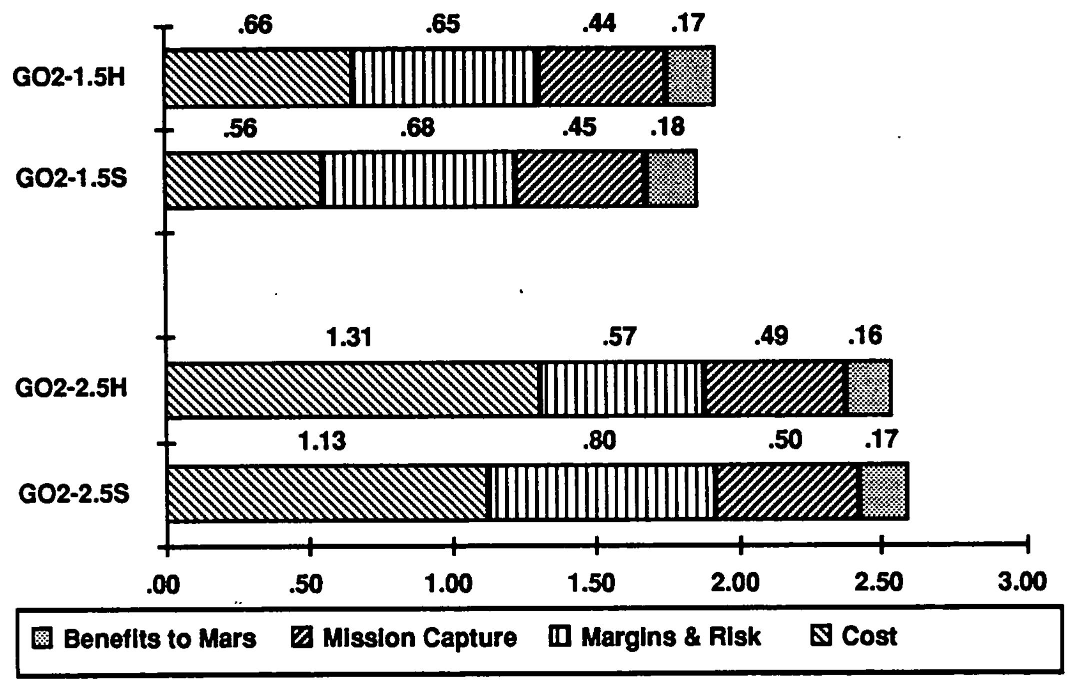




NOTE: Low scores are best

Figure 2-1.1.4-4. Number of Stages Trade Scores (Sheet 2 of 2)

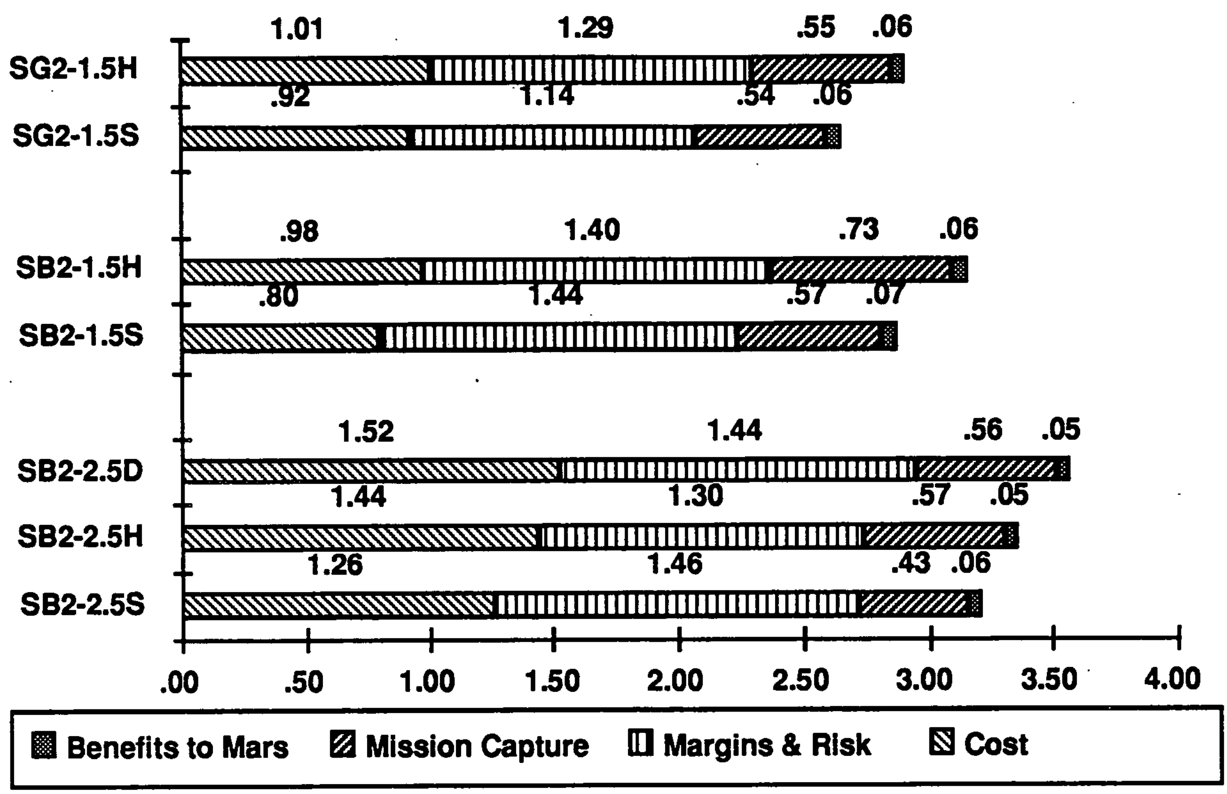




NOTE: Low scores are best

- **Hybrid and Dual crew modules valid with use of LLO for mass drop-off only**

Figure 2-1.1.4-5. Crew Modules Trade Scores (Sheet 1 of 2)



NOTE: Low scores are best

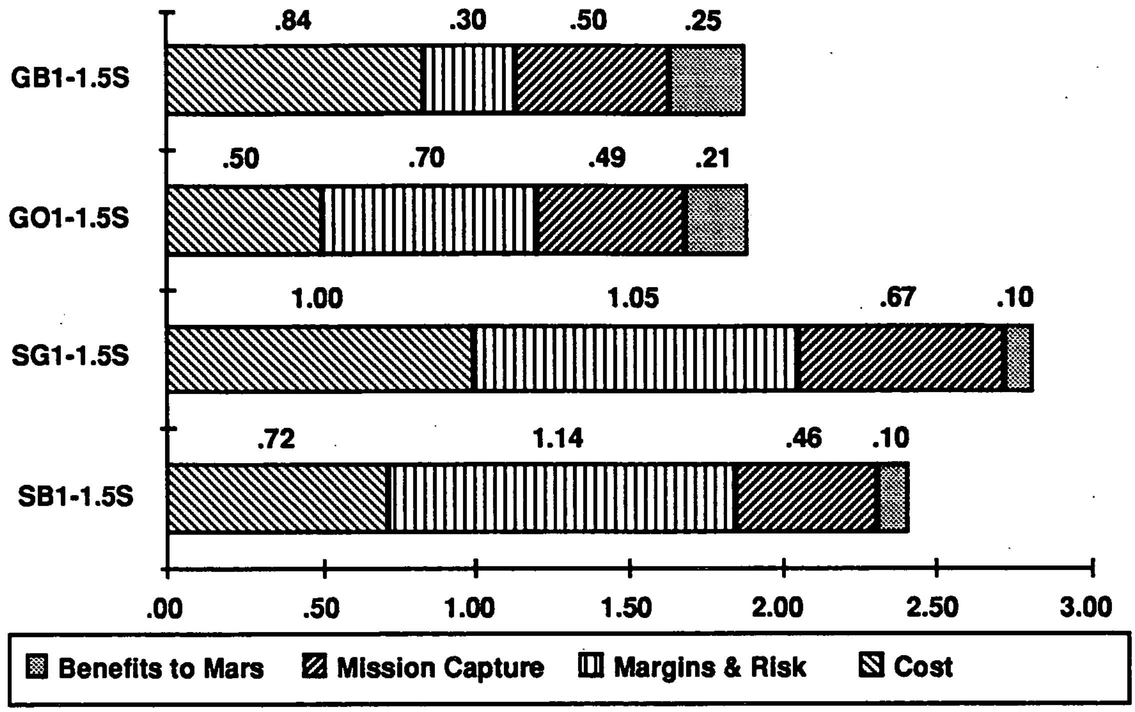
• Hybrid and Dual crew modules valid with use of LLO for mass drop-off only

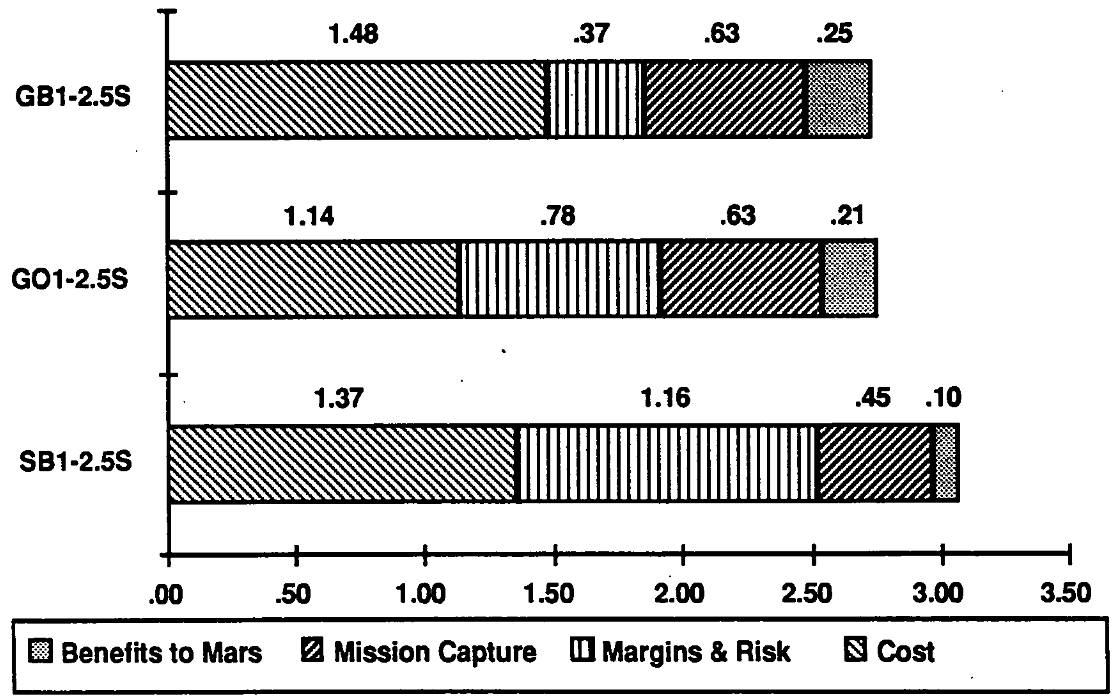
Figure 2-1.1.4-5. Crew Modules Trade Scores (Sheet 2 of 2)

and hybrid crew module approaches were close, with the dual crew module losing. In general, the single crew modules had the lowest cost with the hybrid crew modules having less risk due to the presence of two independently pressurized volumes available for the majority of the mission. The dual crew modules had the highest costs due to the LLO basing of the crew module along with the higher costs associated with development of two elements. Note that the hybrid and dual crew modules were options only when a LLO node was used for mass storage during the missions (i.e., LOR lunar approach trajectory option). Based on the generally better scores for the single crew module, along with the results of the lunar approach trajectory trade, the single crew module was selected.

CAMUS Incorporated, a consulting company formed by William Pogue and Gerry Carr, which was under subcontract for this study, assessed the crew module options from a safety and abort perspective. Their assessment, in summary, was that: the single crew module was preferred for operational simplicity. Also, undesirable risk was introduced by the other crew module options, which required rendezvous and docking, possible long storage periods in orbit, and on-orbit mating of multiple interfaces.

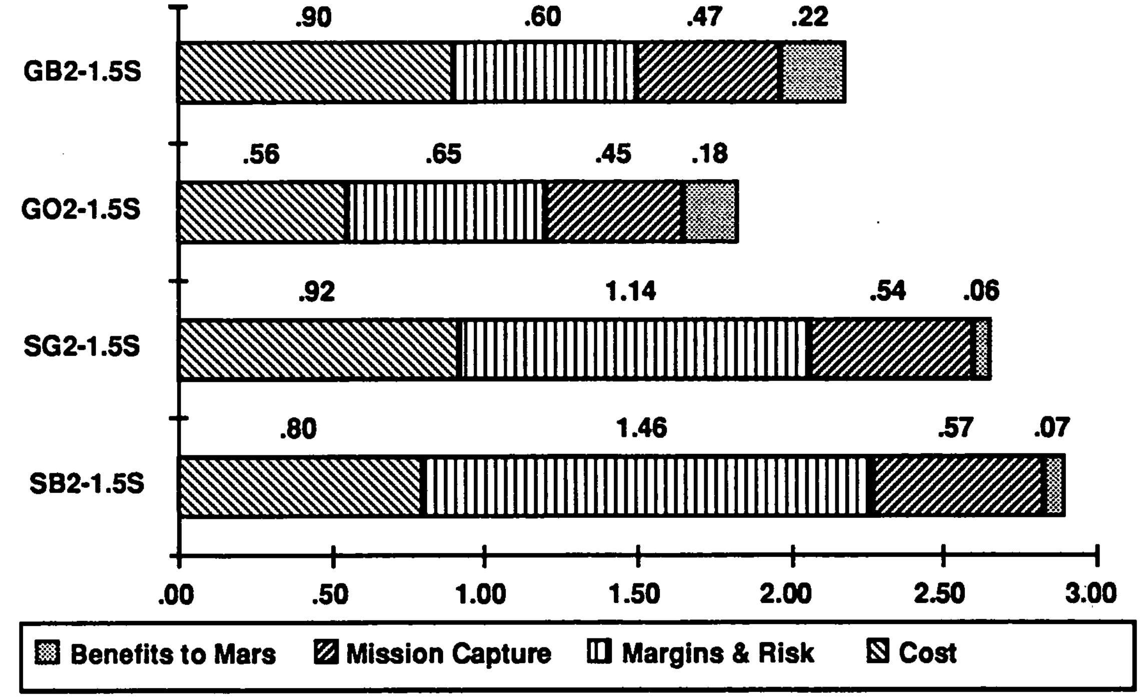
Basing Location. The scoring for the scenarios used to compare basing approaches is shown in Figure 2-1.1.4-6. In general, the two ground-based options scored better than the two space-based options based on generally lower costs and reduced risks. The GO option scored best on costs as all refurbishment operations took place on the ground. GB also had ground refurbishment; however, this option incurred a $7 billion penalty for development of the large booster (approximately 250 metric tons). The lunar and Mars missions were seen as the only missions benefiting from this size booster with the lunar missions having the initial requirement and thus a share of the development costs (primarily facilities modifications). In options where the LOR approach was used, the combination space/ground based vehicle had better scores than the space-based vehicle; however, for the 1.5-stage, single crew module vehicle that did not use LOR (all approaches that were selected in other trades), the space-based scenario had a better score than the combination-based scenario. Note that the space-based scenarios, nominally based at Space Station Freedom, incurred a $4.5 billion cost for modifications

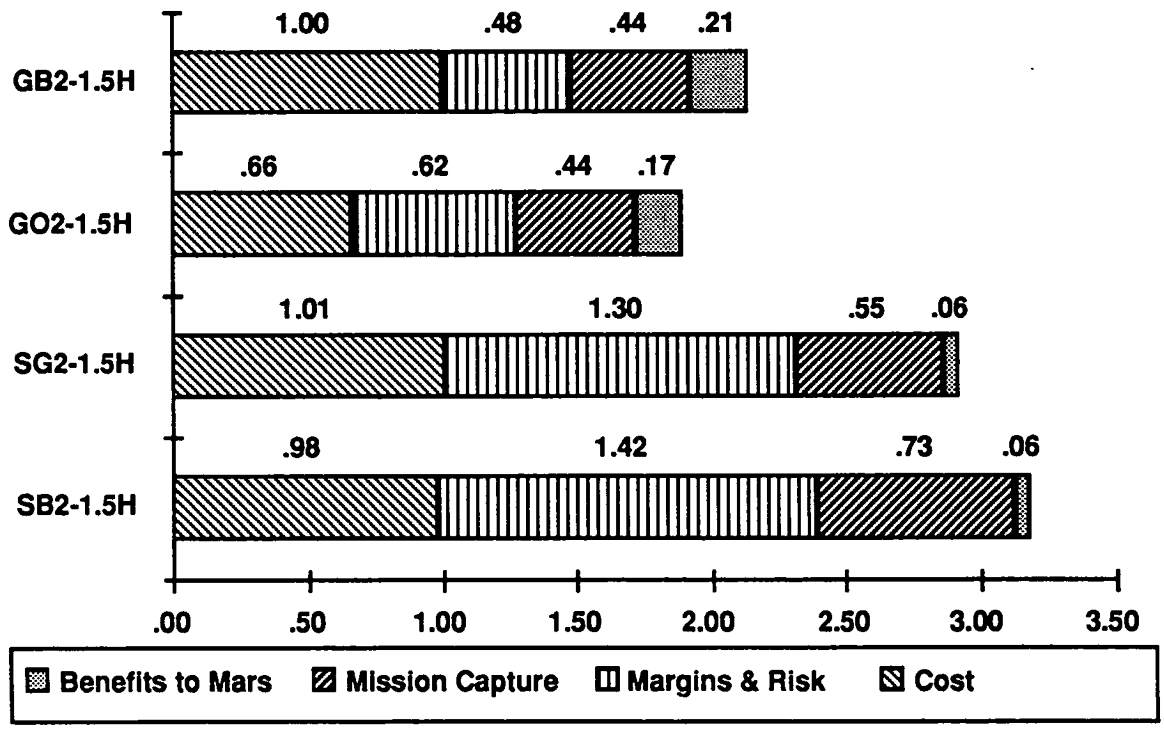




NOTE: Low scores are best

Figure 2-1.1.4-6. Basing Trade Scores (Sheet 1 of 3)





NOTE: Low scores are best

Figure 2-1.1.4-6. Basing Trade Scores (Sheet 2 of 3)

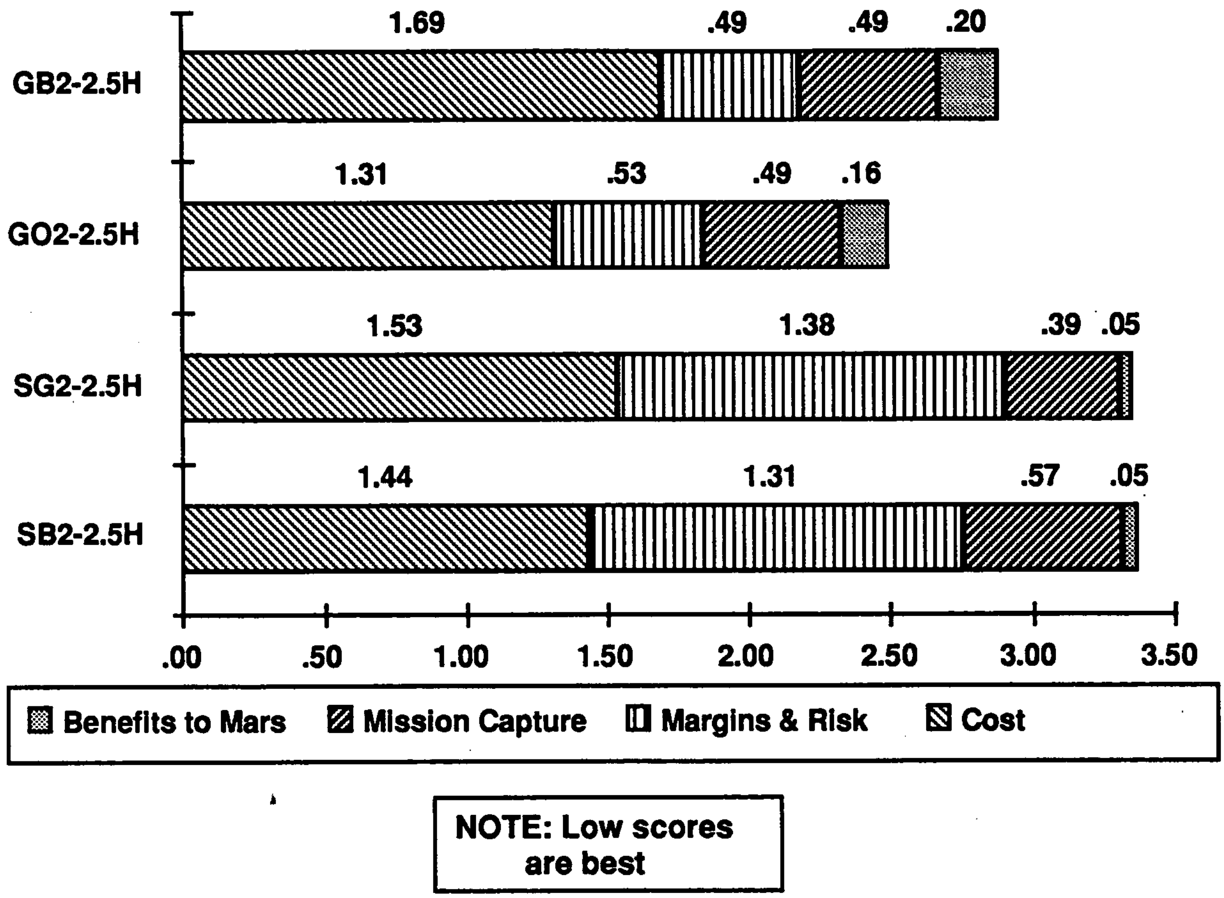


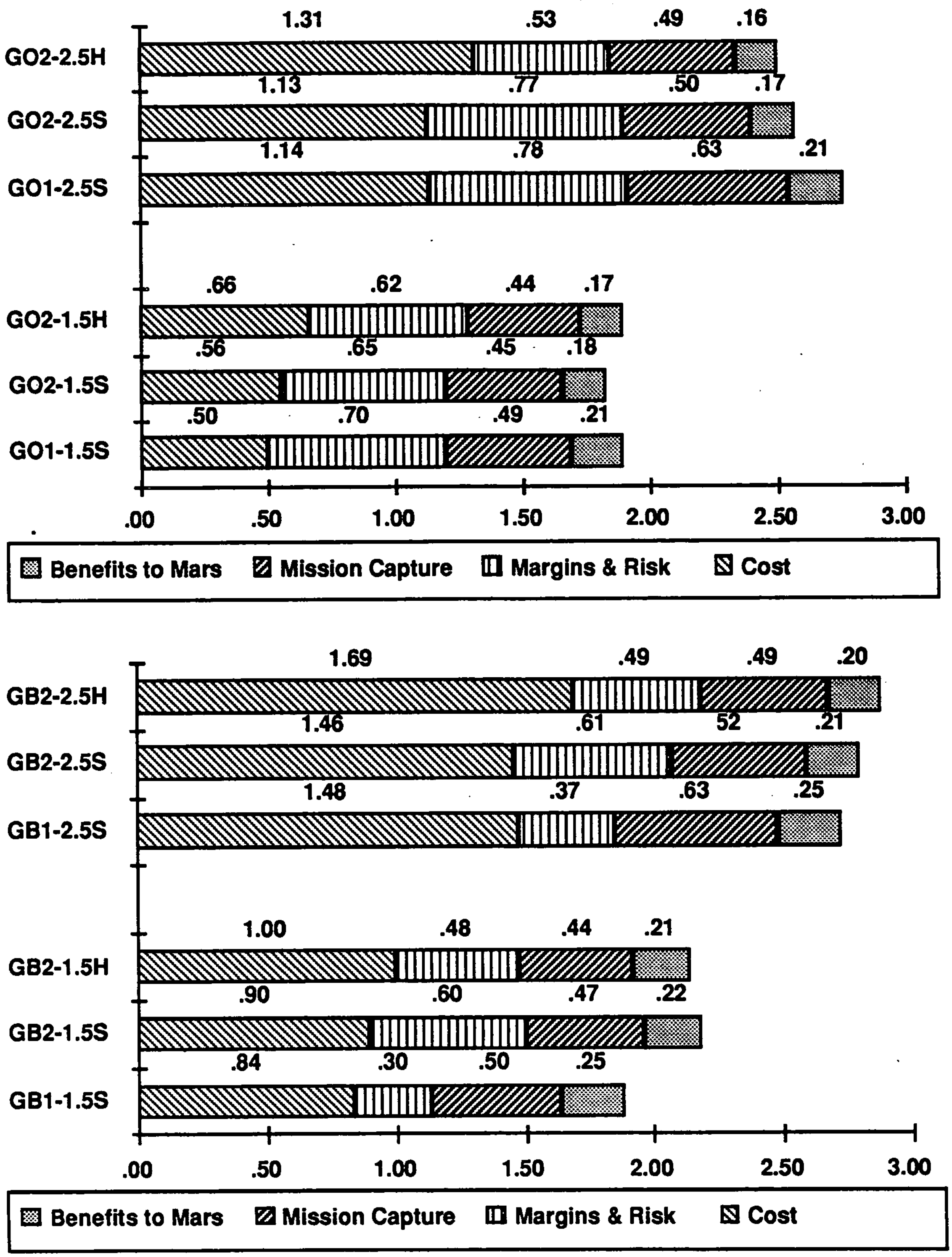
Figure 2-1.1.4-6. Basing Trade Scores (Sheet 3 of 3)

to the SSF. This cost estimate, taken from the General Dynamics STIS, was broken down into top-level elements and examined and accepted as a reasonable estimate.

One of the intentions of the study was to develop and provide a decision database with basing being seen as a primary issue in the definition of the STV. For these reasons, the three basing options were retained in the downselected scenarios to allow more detailed definition of the impacts and costs of the different basing approaches. The different basing approaches depend, in many respects, on other space transportation infrastructure considerations. For example, the GB concept requires booster capability on the order of 260 metric tons, the GO concept requires booster capability on the order of 125 metric tons, and the SB concept requires a 71 metric ton booster. By carrying the three options, a database is available in response to other infrastructure decisions. An examination of the top ten scores reveals that, if the LOR approach is not used, the top three scenarios were selected for further definition.

Lunar Approach Trajectory. At the time the System Architecture Trade Study was being conducted, only two lunar approach options had been identified. These two were the lunar direct approach with a single burn landing and the LOR approach, similar to Apollo, with a lunar orbit insertion burn, mass being dropped off in orbit, and transfer orbit insertion and landing burns. One of the groundrules for LOR was that, if the STV stopped in LLO on the way to the surface, LLO would also be used on the return trip. The idea here was that LLO was being used for storage of elements (e.g., stages, propellant, aerobrake, and crew modules) required for return. If elements were only to be jettisoned, then use of LLO was not necessary.

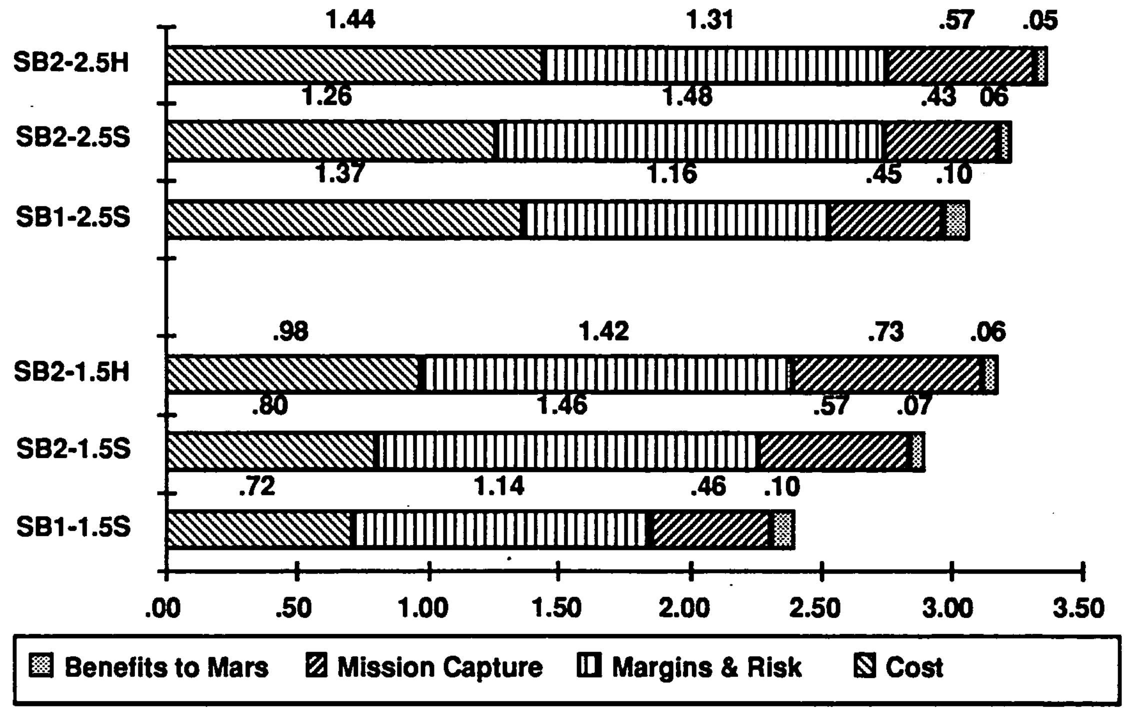
After the trade was nearly complete, the two-burn lunar orbit direct (LOD) approach was identified. An assessment of this approach showed that, in terms of the evaluation criteria used, LOD was similar to the direct approach. The direct approach was seen to be preferred over the LOR approach and the differences between the direct approach and LOD only favored LOD. For example, the LOD approach had better performance than the direct approach (lower overall ETO costs). Figure 2-1.1.4-7 shows the scores for the LOR versus

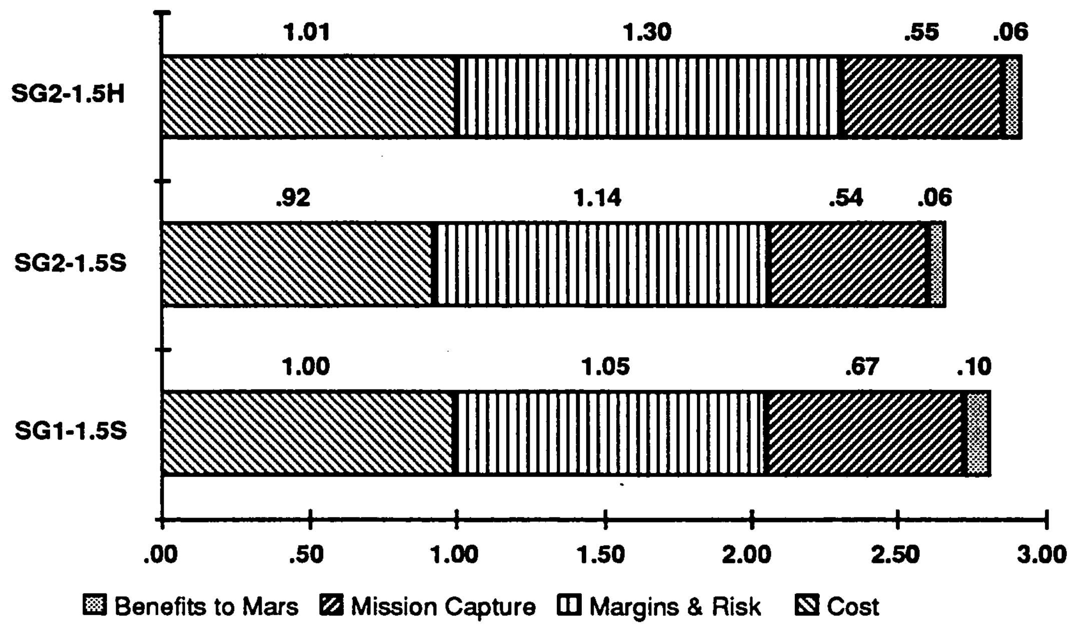


NOTE: Low scores are best

• Only LLO and LS Direct compared

Figure 2-1.1.4-7. LLO Versus LS Direct Trade Scores (Sheet 1 of 2)





NOTE: Low scores are best
• Only LLO and LS Direct compared

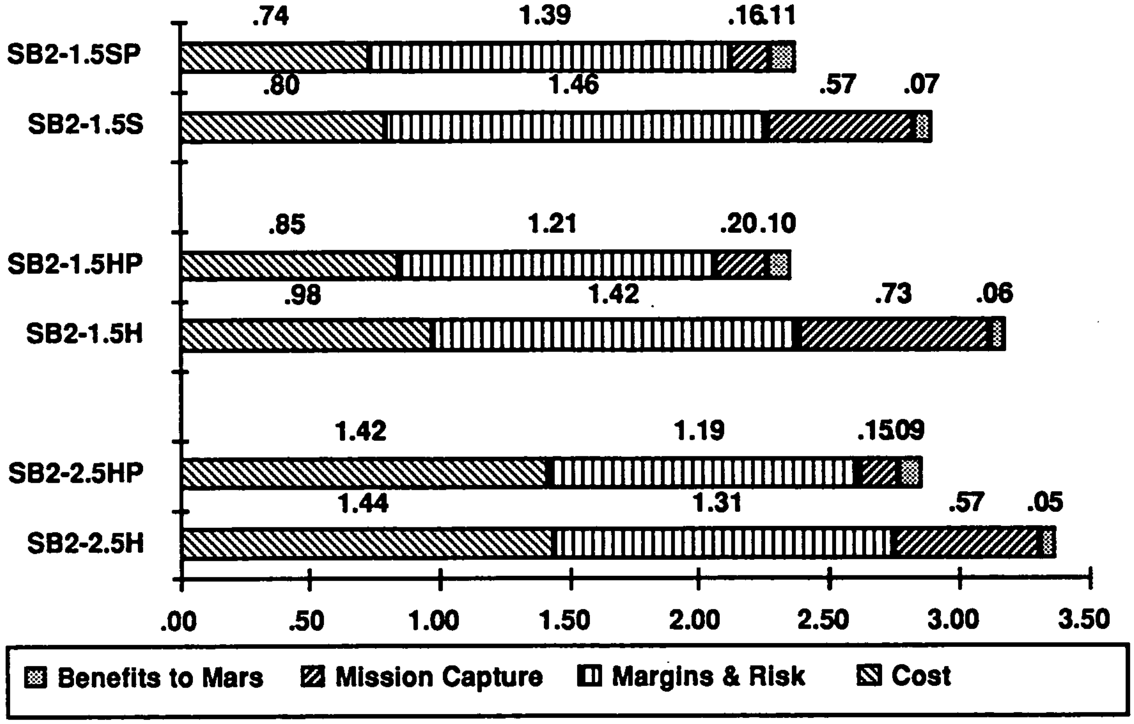
Figure 2-1.1.4-7. LLO Versus LS Direct Trade Scores (Sheet 2 of 2)

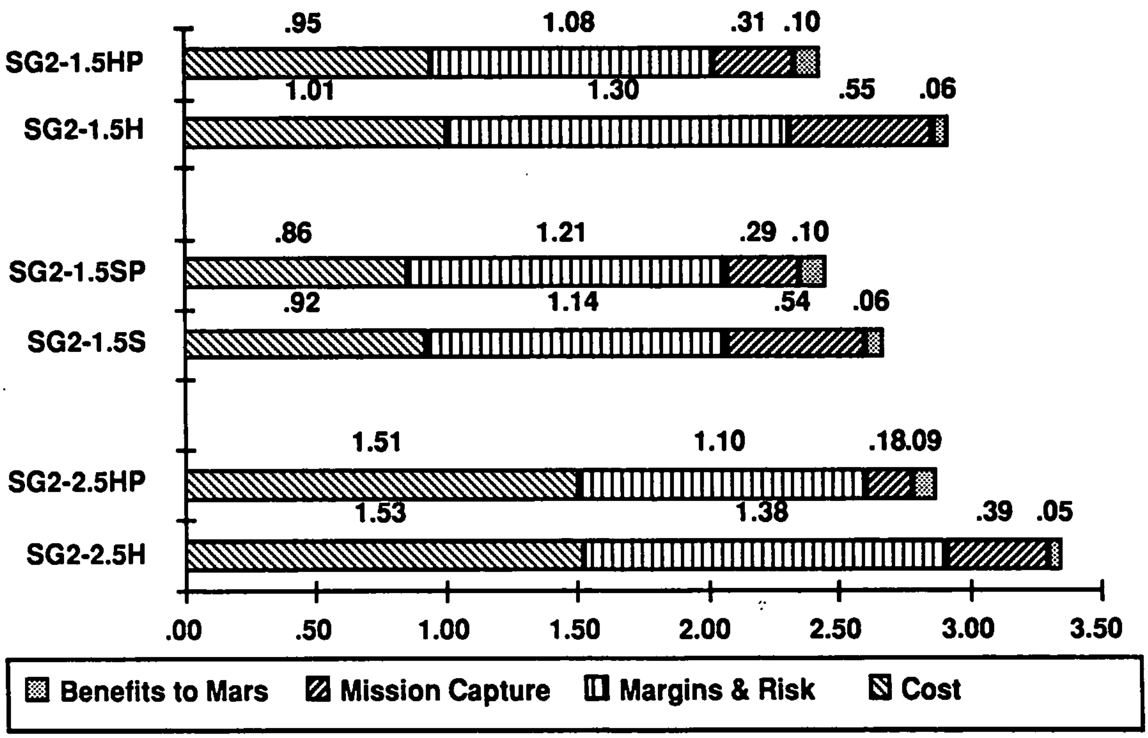
direct evaluations. Note that the direct approach won for six out of seven of the scenario groupings used to evaluate this trade.

CAMUS was also asked to assess the lunar approach trajectory options in terms of safety and abort considerations. Their assessment was that (1) the LOD approach appears feasible and worth pursuing, (2) initial use of the fractional orbit approach may be optimistic and the initial use of multiple orbits with growth to the fractional orbit approach may be desirable, (3) leaving elements required for Earth return in LLO for up to 6 months during the missions (LOR approach) introduces risk and is not the preferred approach, (4) the LOD approach builds on instead of duplicating Apollo experience, (5) and if a multiple orbit LOD scenario is initially selected, accommodations for growth to the fractional orbit approach should be guaranteed (i.e., not precluded by configuration, propulsion, and so forth).

Based on the trade results and the CAMUS assessment, LOD was selected as the lunar approach trajectory for the downselected scenarios. At this time, the terminology used to identify the scenarios was modified to delete the reference to the direct versus LOR approaches (i.e., SB1-1.5S became SB-1.5S).

Aerobraked Versus All-Propulsive Return. Six scenario pairs, all using LOR, were initially used to trade the return options. Another pair (space-based, 1.5-stages, single crew module) was then added where the direct trajectory approach was used (Figure 2-1.1.4-8). Cost slightly favored all-propulsive, influenced both by the 70% DDT&E component of the cost scoring, effectively penalizing the aerobrake, and the low-boost cost of $1,000/kg favoring the all-propulsive approach with the required additional propellant available in LEO at a relatively low cost. Margins and risk, somewhat obviously, also favored the all-propulsive approach as this type of operation has been done before where as use of the aerobrake would entail an all new development. The benefits to Mars criterion favored, again somewhat obviously, the aerobrake approach as aerobraking is required for a Mars landing. Note that the margins and risk and benefits to Mars criteria tended in opposite directions as new technology and operational approaches obviously entail a higher level of risk than use of existing hardware and operational concepts. The relative weighting of the criteria was an important factor in the all-propulsive approach having the best





NOTE: Low scores are best

Figure 2-1.1.4-8. Aerobrake Versus All-Propulsive Trade Scores (Sheet 1 of 2)

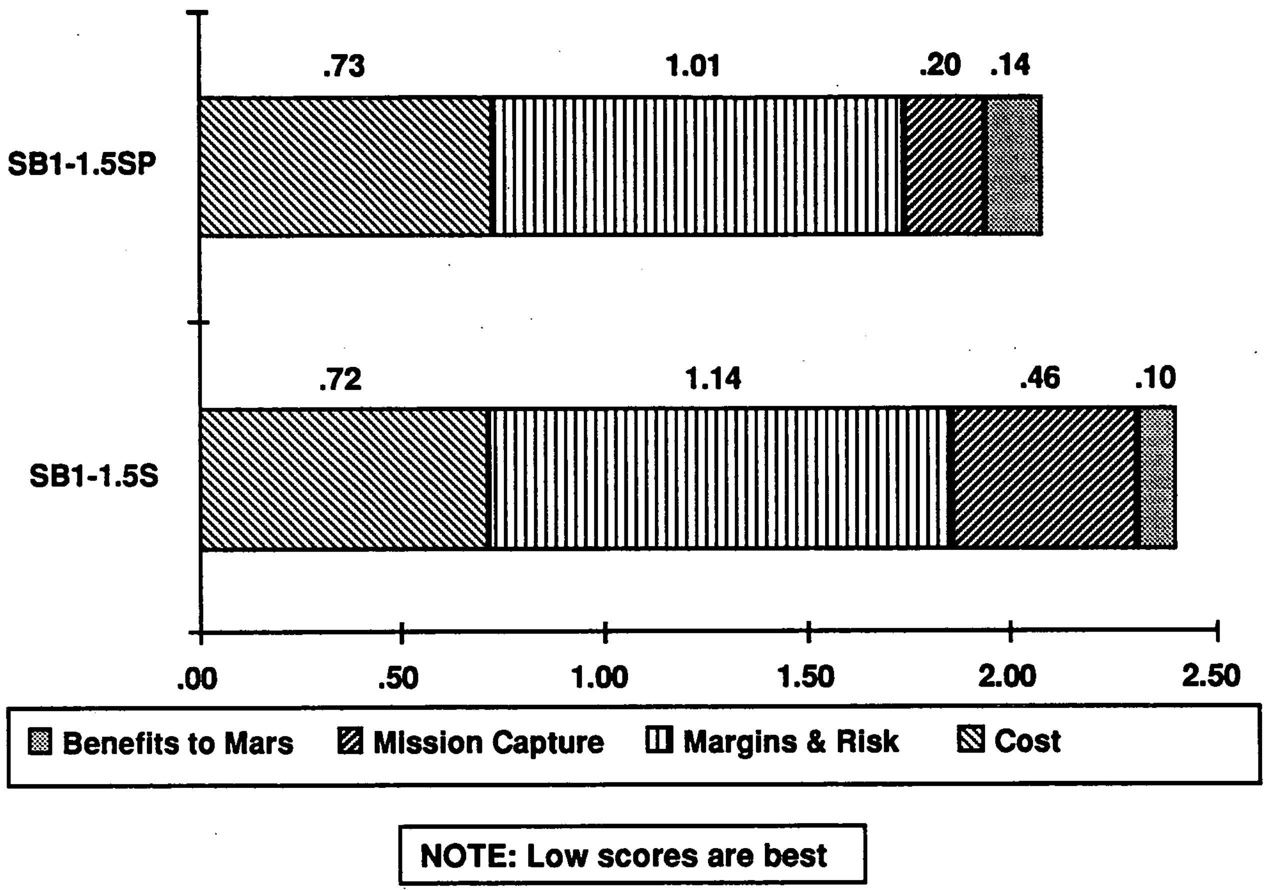


Figure 2-1.1.4-8. Aerobrake Versus All-Propulsive Trade Scores (Sheet 2 of 2)

scores. Section 2-1.1.5 looks at sensitivities to the evaluation criteria for this trade.

The aerobrake was retained in the interest of developing the technical database and aerobrake details. Additionally, the evaluation methodology did not allow for higher weighted scoring based on mitigating factors. In this case, the lunar transportation system is the only SEI opportunity to prove out aerobraking, unlike other technology/operational areas that will benefit from development and operation of the SSF, lunar base, and new ETO systems.

Droptanks Versus Propellant Tankers. Two scenario pairs were used to trade the use of droptanks versus propellant tankers. Figure 2-1.1.4-9 shows the scoring for this trade. Note that this trade (along with the entire system architecture trade) was based on the lunar missions only, with the exception of the mission capture evaluation, which used the lunar transportation system optimized elements as required to perform the non-lunar DRMs. Based on the lunar missions, the use of droptanks was slightly favored over the use of propellant tankers and was selected as the baseline for the space-based vehicles.

2-1.1.5 System Architecture Trade Study Sensitivities

Architecture trade study sensitivities were examined for the aerobrake versus all-propulsive trade. The effects of varying the weighting of evaluation criteria, varying the cost score components of DDT&E and LCC, and varying boost cost were examined. For all of these sensitivity evaluations, the identical scenario was used with the only variation occurring in the use of main propulsion versus the use of the aerobrake to return to the LEO node. The scenarios used were both space based with a single stage using TLI and lunar descent droptanks, single crew modules, a direct to the lunar surface trajectory, and in one case an all-propulsive return (SB1-1.5SP). In the other case, an aerobraked return was used (SB1-1.5S).

Effects of Varying Evaluation Criteria Weighting. The sensitivity of the aerobrake versus all-propulsive trade study to the criteria weighting is shown in Figure 2-1.1.5-1. Note that the cost score consisted of 70% DDT&E, 30% LCC

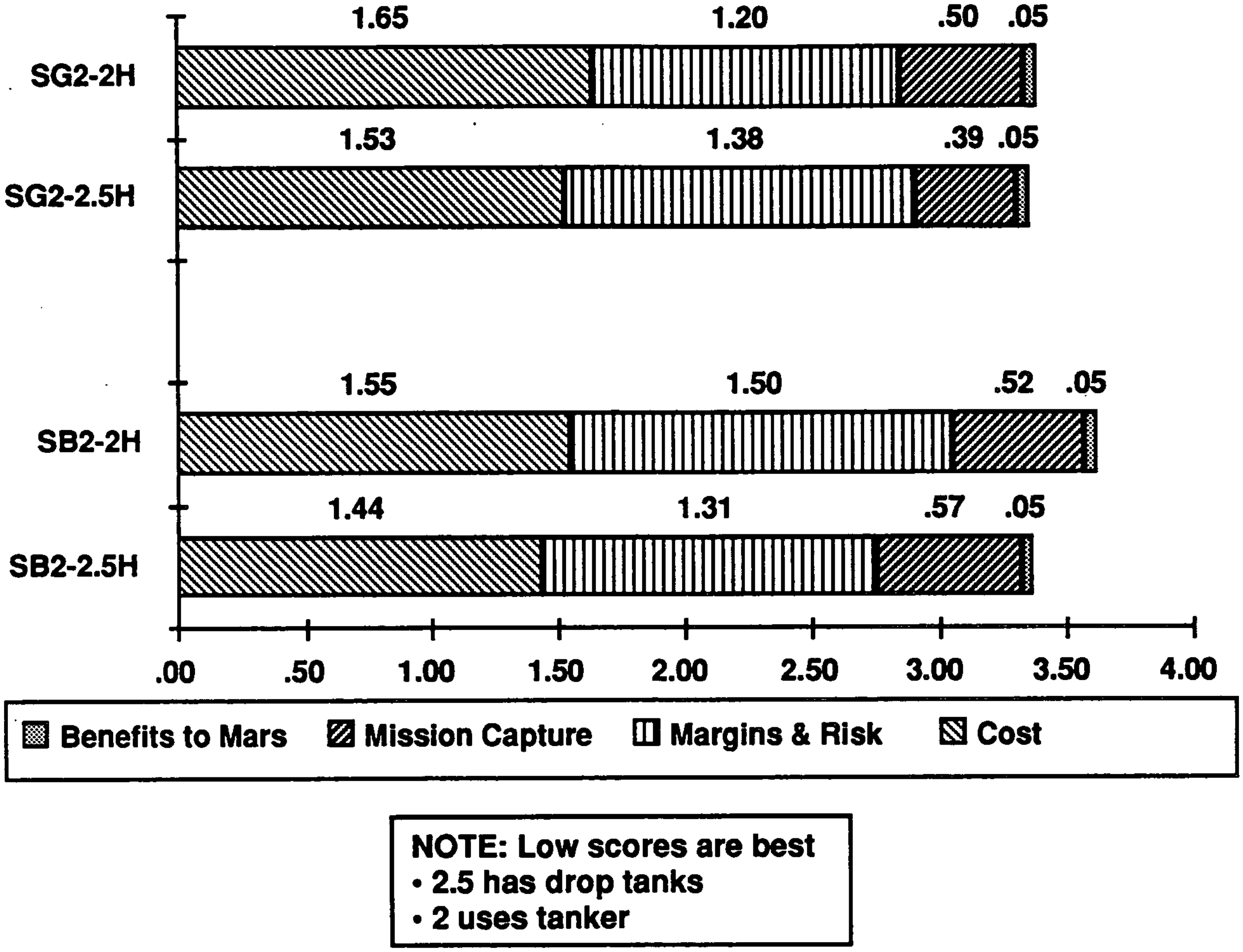


Figure 2-1.1.4-9. Droptanks Versus Tankers Trade Scores

with LCC consisting of DDT&E, recurring costs (i.e., hardware production and operations), and boost or ETO costs (based on low-boost costs of $1,000/kg or $454.5/lb). Sensitivities to differences in cost score make-up are addressed later.

From left to right the criteria mix consisted of 100% cost; 50% cost, 30% margins and risk, 15% mission capture, 5% Mars benefits (the criteria mix used in the study); and then a successive 50% weighting of each of the remaining criteria with, in each case, the additional criteria being given their relative weighting percentage as used in the study. For example, in the margins and risk driven case, margins and risk is given a 50% weighting. From the weighting used in the study (50% cost, (30% margins and risk), 15% mission capture, 5% Mars benefits), the remaining total of weighting percentage is 70% (50% + 15% + 5%). Cost is then weighted at 50% of the 70%, mission capture is weighted at 15% of the 70%, and Mars benefits is weighted at 5% of the 70%. Together with the 50% for margins and risk, the total is equal to 100%. Margins and risk is emphasized and each of the other criteria keep their relative weighting with respect to each other. The same procedure was used to emphasize the mission capture and Mars benefits criteria.

What can be seen from the results is that a strong emphasis on either cost or benefits to Mars criteria favors the aerobrake. Both margins and risk and mission capture favor the all-propulsive case. Margins and risk and mission capture together favor all-propulsive more strongly than cost together with Mars benefits favors the aerobrake. This can be seen in the point for the criteria weighting used in the study. In this case, cost and Mars benefits together is equal to 55% of the score and margins and risk together with Mars benefits is equal to 45% of the score. In this example, the all-propulsive approach wins so the margins and risk plus mission capture favors all-propulsive more strongly than cost plus Mars benefits favors the aerobrake.

Effects of Varying Cost Score Components. The cost scores used in the System Architecture Trade Studies were composed of 70% DDT&E and 30% LCC. LCC comprises DDT&E, recurring costs (hardware production and operations), and ETO costs. Thus, DDT&E actually made up somewhat more than 70% of the cost score. The rationale behind this make-up of cost elements

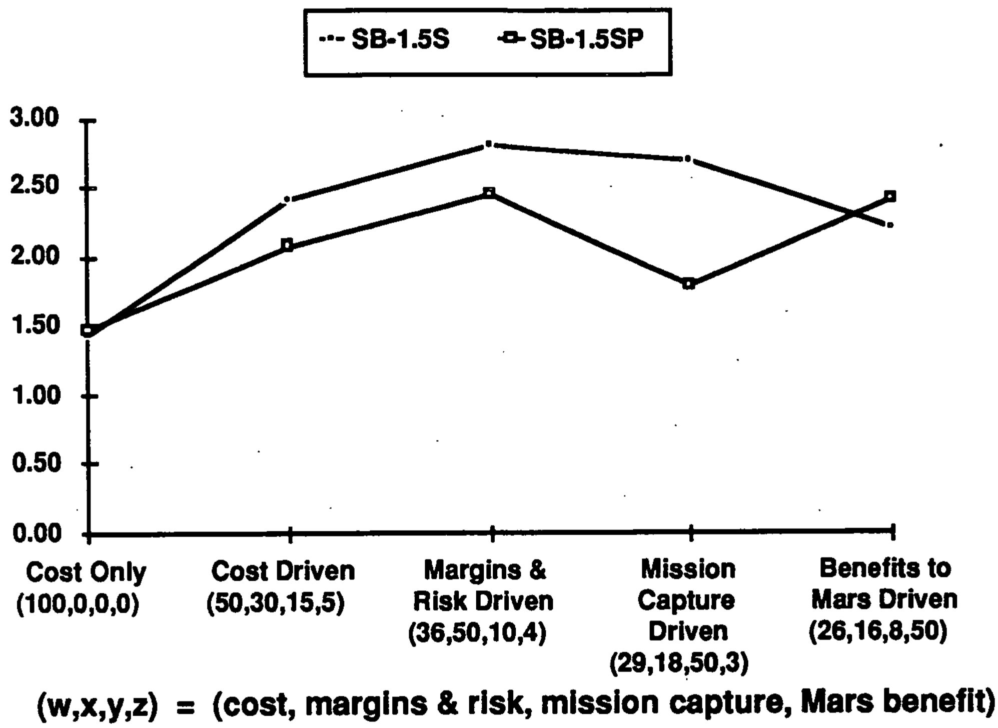


Figure 2-1.1.5-1. Effect of Varying Criteria Weighting

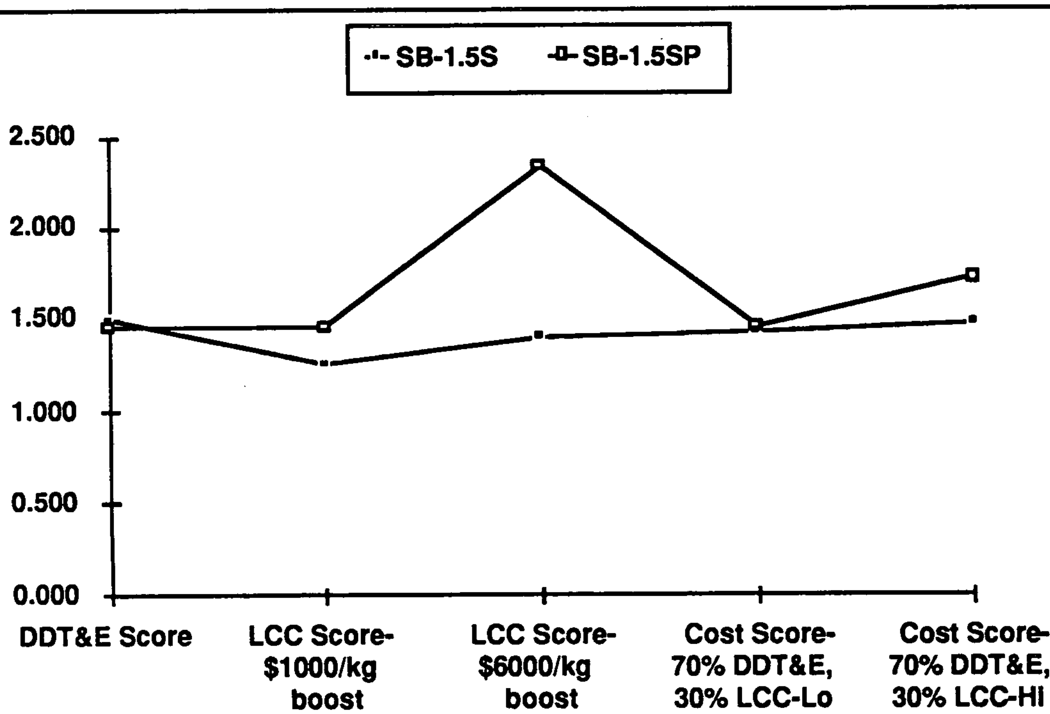


Figure 2-1.1.5-2. Effect of Varying Cost Components on Cost Score

in the cost score is that DDT&E comprises the funding for the initial years of a program. Historically the front-end funding profile has been a significant factor in Government funding approval for a new program.

Figure 2-1.1.5-2 presents the cost scores as a function of make-up and weighting of the different cost elements. DDT&E only favors the scenarios using the all-propulsive approach because DDT&E is higher for the aerobrake than for the all-propulsively returned scenarios. LCC alone favors the aerobrake, especially if the high-boost cost is used. When the cost score comprises 70% DDT&E and 30% LCC, with the LCC based on low-boost cost, the cost scores are quite close between the aerobraked and all-propulsively returned scenarios with the aerobrake slightly preferred. Use of the high-boost cost in the LCC with a 70% DDT&E and 30% LCC cost score more strongly favors the aerobrake. The following discussions and figures look at the effects of variations in criteria weighting when different cost scoring approaches are used.

Using a cost score of 70% DDT&E and 30% LCC (based on the low-boost cost of $1,000/kg), the weighing of the cost score was varied from 100% down to 50% with the other criteria keeping their relative weighting (i.e., margins and risk is twice as important as mission capture and six times as important as benefits to Mars). As can be seen in Figure 2-1.1.5-3, cost favors the aerobrake; however, when the low-boost cost is used the aerobrake is favored only if cost is weighted in the 97% range.

When the high-boost cost ($6,000/kg or approximately $2,750/lb) is used in a 70% DDT&E and 30% LCC cost score (Figure 2-1.1.5-4), the aerobrake is favored if cost is weighted in the 74% range; again, all other weighting remaining in relative percentage. The all-propulsive case requires more propellant and thus when ETO costs are higher, the all-propulsive case begins to appear less attractive. However, in the preceding two cases with a 70% DDT&E and 30% LCC weighting, the ETO costs are a relatively small part of the overall cost score so the effects seen by varying the boost costs are not emphasized.

Figure 2-1.1.5-5 contains the results of a variation in evaluation criteria weighting when the cost score uses LCC only, in this case with the low-boost

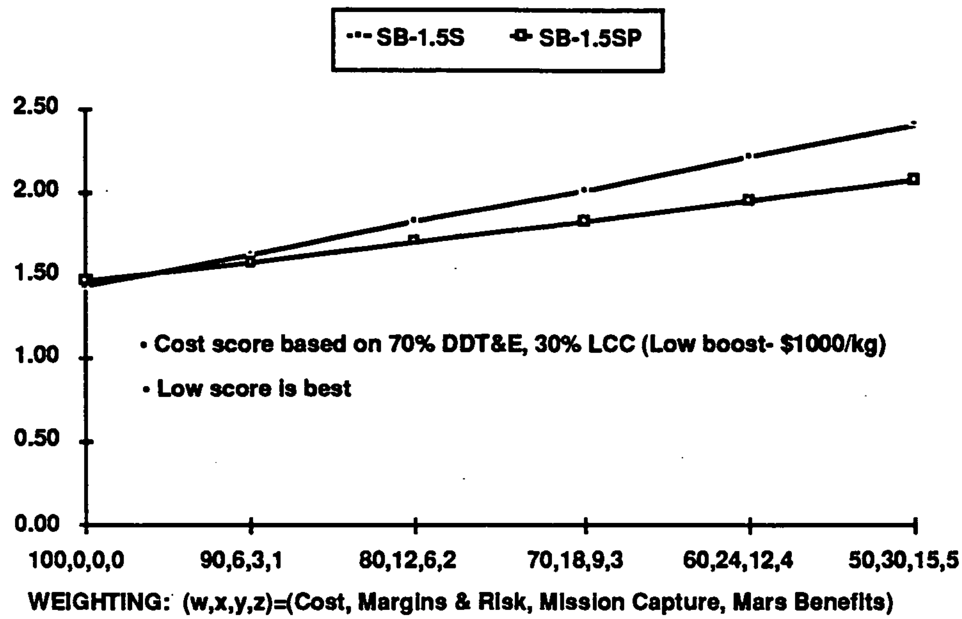


Figure 2-1.1.5-3. Cost Score Weighting Sensitivity - 70% DDT&E and 30% LCC (Lo)

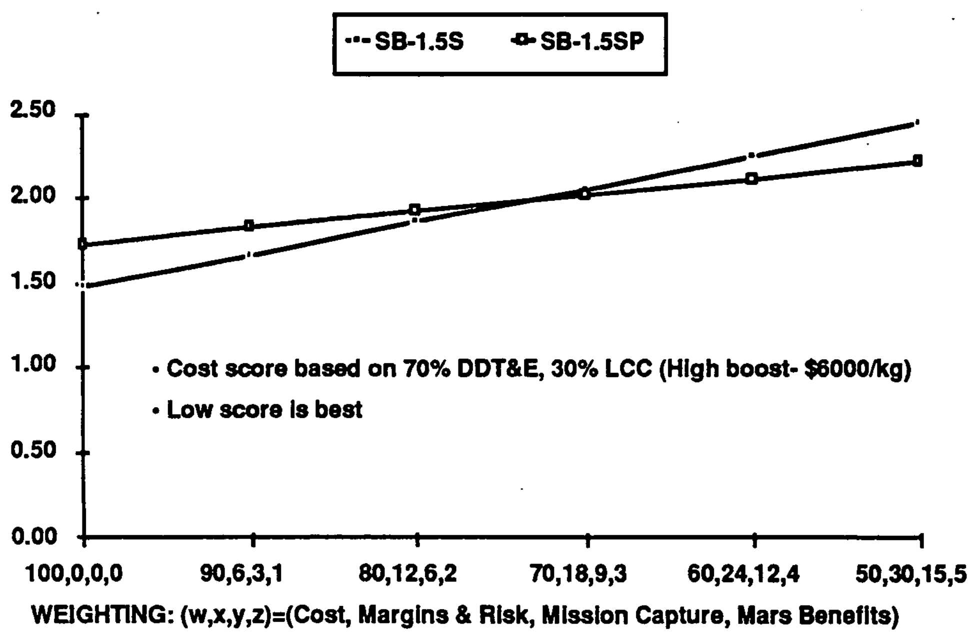


Figure 2-1.1.5-4. Cost Score Weighting Sensitivity - 70% DDT&E and 30% LCC (Hi)

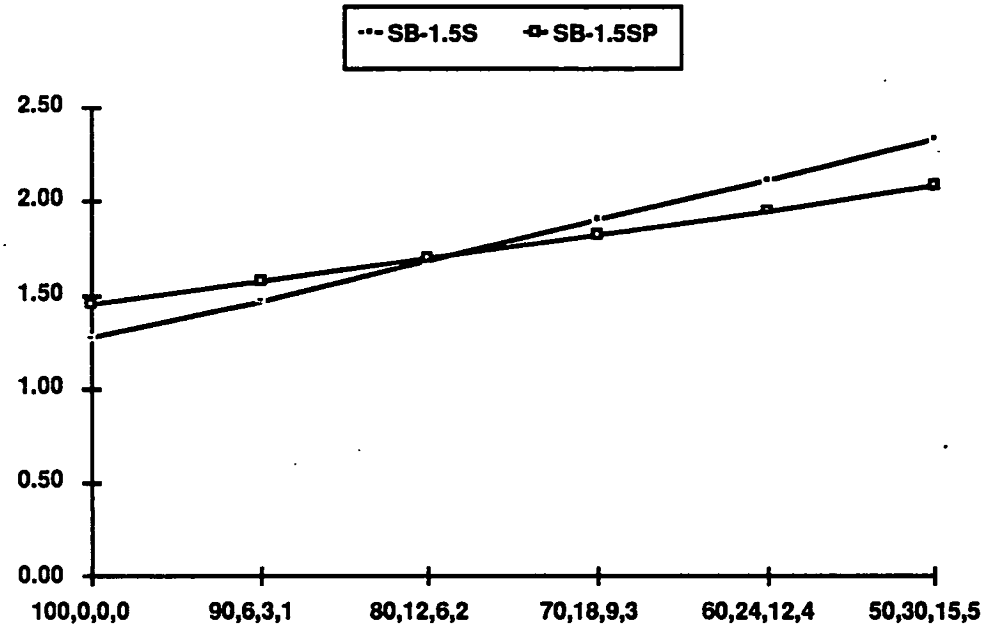


Figure 2-1.1.5-5. Cost Score Weighting Sensitivity - 100% LCC (Lo Boost)

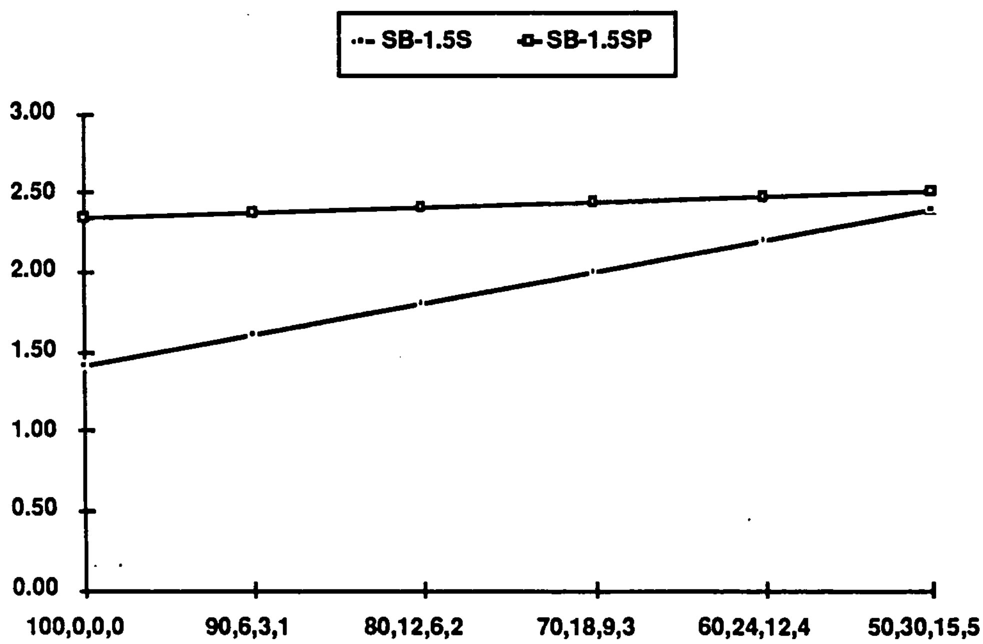


Figure 2-1.1.5-6. Cost Score Weighting Sensitivity - 100% LCC (Lo Boost)

cost score. By not emphasizing DDT&E, the relative importance of boost costs is increased. When cost is weighted in the 78% range, the aerobrake wins (compared to winning in the 97% range when using 70% DDT&E and 30% LCC low boost). In Figure 2-1.1.5-6, only LCC is again used as the cost score but this time with the LCC based on the high-boost cost. As the relative emphasis of boost cost is increased, higher boost costs swing the selection to the aerobrake when the cost weighting is in the 50% range (as compared to the 74% range when 70% DDT&E, 30% LCC high boost is used).

Effects of Varying Margins and Risk and Mars Benefits Weighting. The margins and risk and Mars benefits criteria are opposed to each other as, somewhat obviously, the introduction of new technologies and operations that benefit the Mars program results in higher risk. In Figure 2-1.1.5-7, the sensitivity of the importance placed on the two criteria is shown. When the two criteria are equally weighted (in fact up to 65% margins and risk and 35% Mars benefits), the aerobrake wins.

2-1.2 STV EVOLUTION

One of the objectives of the STV study was to provide an evolvable transportation system. When one examines the history of the space program, especially launch vehicles, it becomes apparent that vehicles will evolve to meet the mission requirements. If a vehicle program starts with this expectation, as do airplane programs, then the changes can be accommodated more efficiently.

With planning, overall program risk can be lowered tremendously by using initial vehicles to prove technology before it is absolutely required. Making changes incrementally in a few major building blocks avoids the "all or nothing" approach of bringing a totally new vehicle with all-new systems on line just in time to meet a mission need. With the "all or nothing" approach a slide in any one of the new systems causes the whole program to slide. The alternative is to get something flying early and bring new technology on when it is available. Constraints on the DDT&E program may also be supported by an evolutionary approach. These DDT&E program constraints can be cost constraints on up-front funding, technology constraints based on the lack of maturity of a particular

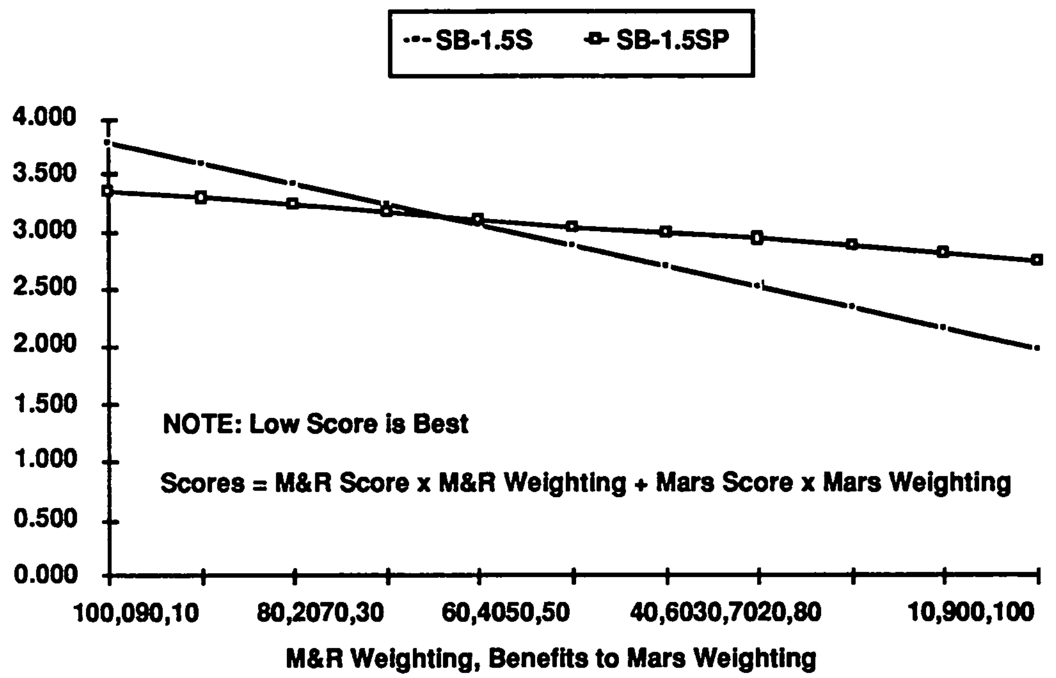


Figure 2-1.1.5-7. Effect of Varying Margins and Risk and Mars Benefits Weighting

technology that is eventually anticipated to provide significant benefits or schedule constraints.

The approach taken in identifying an evolution path for the STV was to identify STV requirements and the timeframe when required capabilities are needed and then develop a time-phased hardware and operations plan. A first cut at a vehicle evolution approach was developed and will be discussed with the note that, as the mission model changes, the evolution plan will change. Development of a viable plan for evolution is an ongoing task. Requirements on the STV derived from current and future missions provided the basic goals and framework for the plan. The more undefined or subject to change these requirements, the more the evolution plan will be subject to change. Additionally, the plan must develop in concert with design and operations trade studies and both direct the design and be responsive to the trade studies. The current state and planned development of desired technology is also a constraint that must be accommodated.

The requirements that provide the framework for evolutionary planning can be divided into those requirements that are absolutely required to perform a mission (enabling) and requirements (enhancing) that support a lower cost of the system (sometimes requiring a higher DDT&E cost) but that are not absolutely required to perform the mission. Figure 2-1.2-1 provides an example of the benefits of a well-defined evolution plan where the technology can be proved before it is required (thus introducing schedule margin) and where the introduction of new technology and operational approaches can be phased in in a manner that provides an acceptable front-end funding profile or buy-in cost.

In this example, the throttleable engine (enabling) is introduced earlier than needed to prove out the engine and gain experience and confidence in its use. Phased in with this is increasing autonomy (enhancing), which entails a front-end DDT&E investment, but should reduce the LCC of the system (and increase reliability) by lowering manpower support requirements. The enabling technology would optimally be phased into the program in a manner that supports a relatively steady ramp-up of the DDT&E budget for an acceptable buy-in cost.

- Requirements can be termed enabling or enhancing
  - Enabling: required to perform the missions, e.g. a throttlable engine is required to accomplish Lunar landing: introduced early for risk reduction
  - Enhancing: supports low cost, e.g. increasing levels of autonomy reduces cost but is not absolutely required to perform the missions: introduced where best fits into mission model AND enhances affordability

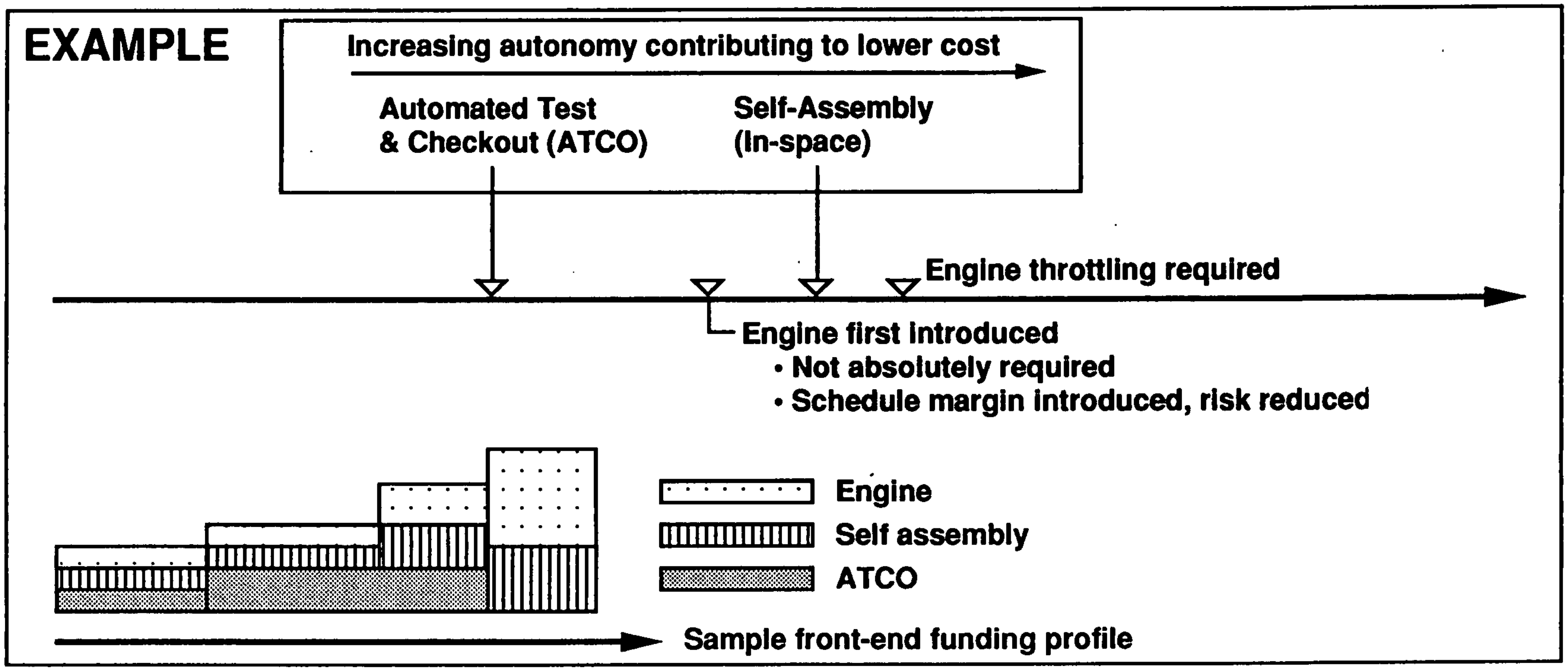


Figure 2-1.2-1. Example of Evolution Benefits

An evolution path is highly dependent on the mission model to provide the current and future vehicle requirements. The current STV mission model is recognized as being preliminary. Figure 2-1.2-2 shows a summary of the time-phased requirements levied on the STV from the design reference missions.

Examination of the phasing of the requirements shows some problems with developing the STV in an evolutionary manner. This mission model requires the majority of the eventual STV capability to be available very quickly after IOC, which does not provide an optimal opportunity for a phased evolution. The LEO tug missions can be seen as being the early drivers for much of the STV capability, such as space basing and reusability. If little time exists between requirements for major capabilities, evolution may not be realistic. Costs associated with the design and implementation of block changes, technology upgrades, and so forth may require that the vehicle be initially developed with virtually full capability.

For example, the throttleable engine is not required until 3 years after startup; however, prior experience with a new or upgraded engine would be desired before committing large lunar cargos or personnel to use of the vehicle. Additionally, initial use of a currently existing engine would force either the new engine to have the same interfaces or require an expensive redesign of the propellant delivery system and engine monitoring to accommodate a change to the new engine. These factors would point toward including the new engine in the initial vehicle. In this particular case, the development schedule may require that a new engine be brought on at a later date, but the sensitivity of vehicle evolution to the mission model requirements is illustrated.

The enabling requirements have been split out in Figure 2-1.2-3 and give the absolute need dates (in accord with the current STV mission model) for the associated capabilities. The enhancing requirements can be separated into autonomy, reusability, and aerobraking. These capabilities can then be incorporated where they best fit into the mission model while supporting a reasonable funding profile.

The evolutionary path identified in Figure 2-1.2-4 provides an early STV to perform the payload delivery and planetary boost missions and then uses

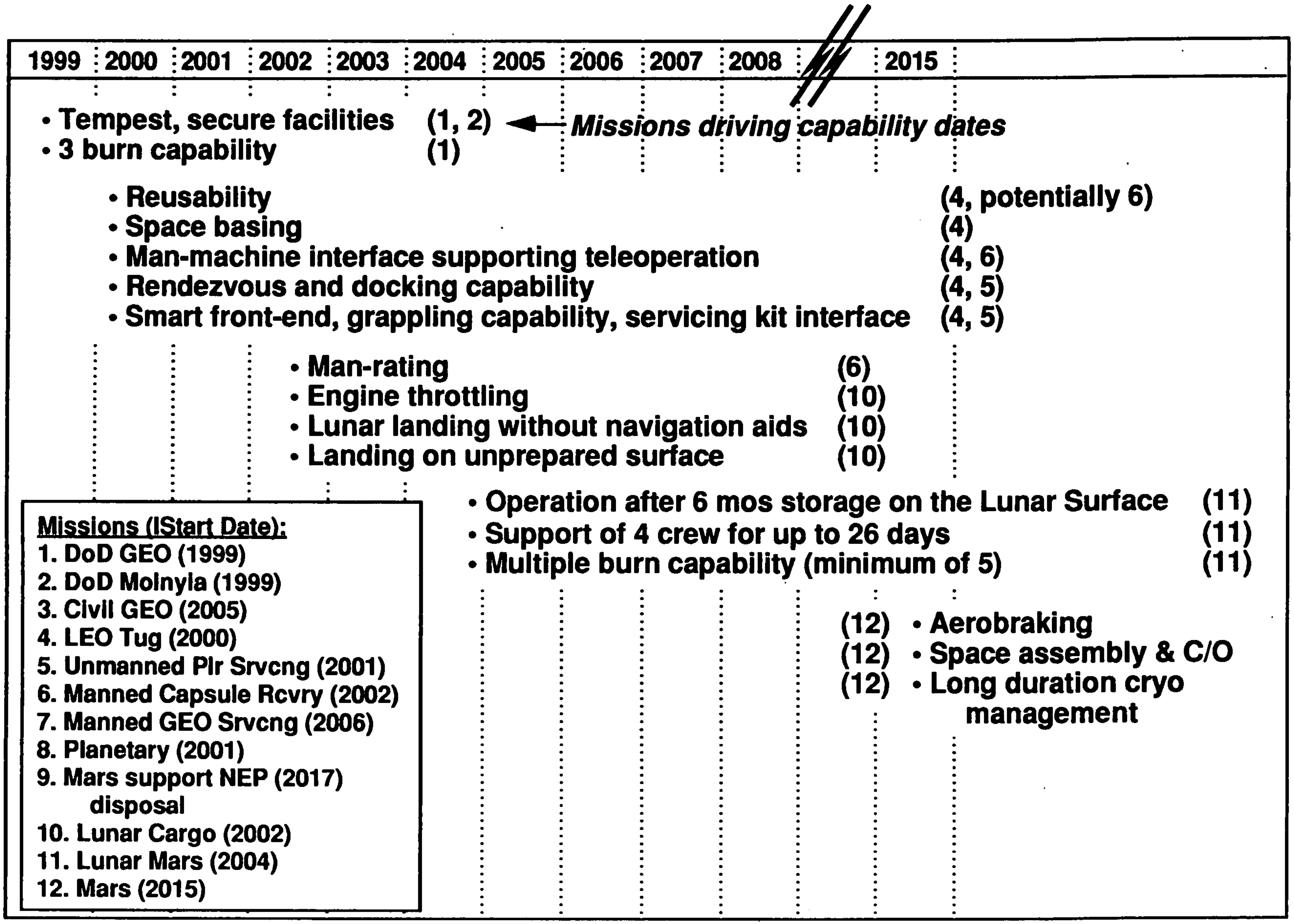


Figure 2-1.2-2. Summary Time-Phased Mission Requirements

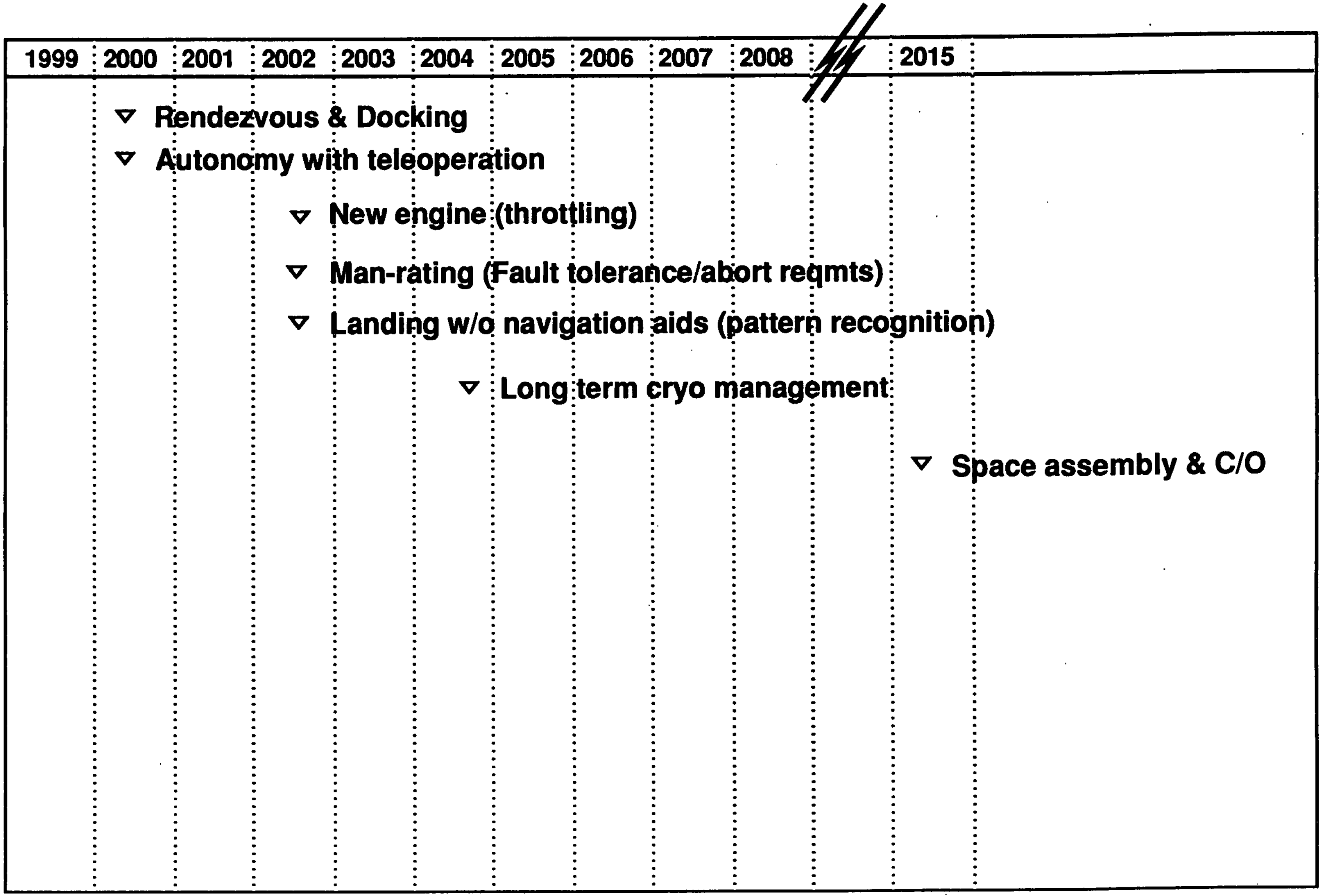


Figure 2-1.2-3. Summary Time-Phased Enabling Requirements

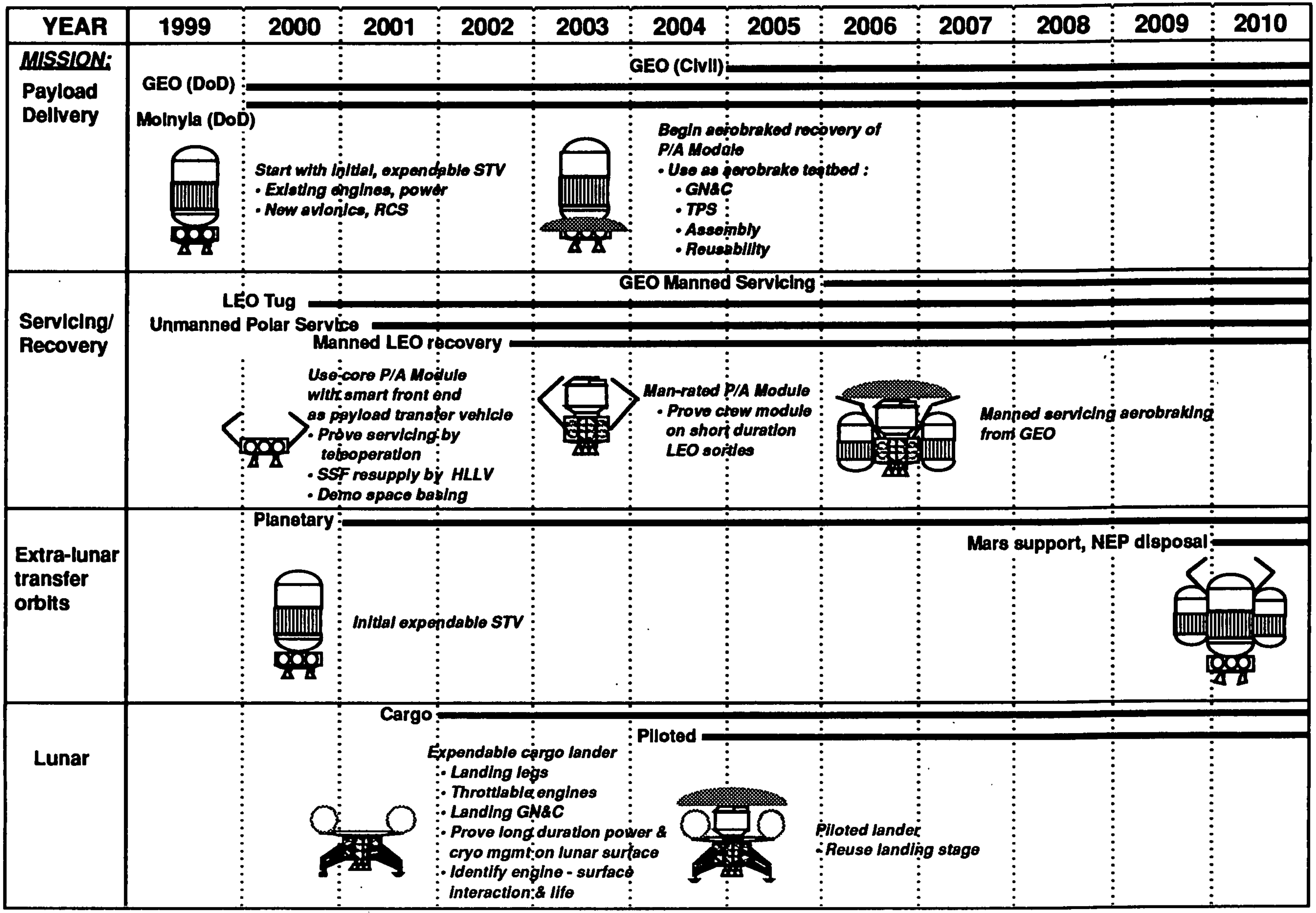


Figure 2-1.2-4. Vehicle Evolution Example

derivatives of this vehicle to prove out needed capability increases (aerobrake and man rating) before absolutely required.

The STV starts initially in an expendable mode, and the space tug missions provide an opportunity to demonstrate space basing and to build up confidence in vehicle reuse. The throttleable engine is brought on in 2002 to support the lunar cargo missions where engine and lunar surface interaction can be investigated prior to piloted flights. These missions will also provide the opportunity to prove out long-duration cryogenic management techniques with the cargo vehicles on the lunar surface. The crew module is brought on in 2002 where it can be used in short-duration LEO missions prior to use in the long-duration lunar missions.

The evolutionary plan shown in Figure 2-1.2-4 provides an example of an evolution plan responsive to the design reference missions. As vehicle trades and detailed design are accomplished and as the mission model is further defined and modified, the evolution plan will be updated to best reduce risk, support the missions, and support funding levels.

2-1.3 SAFETY AND ABORT CONSIDERATIONS

Safety and abort issues were included in the margins and risk evaluation criteria (section 2-1.1.3) in the System Architecture Trade Study. Abort and free return issues will be discussed in this section.

The use of Space Station Freedom, or other LEO nodes, has an impact on orbital operations and the types of missions that are possible. The use of an Earth-orbiting node such as the SSF limits the lunar transfer mission opportunities to every 6 to 11 days instead of daily as is the case for launching from Earth.

This node also limits the Moon return times because the Space Station line of nodes must be nearly in the Earth-Moon plane to keep the ΔV for the Space Station rendezvous low. The ability to conduct a mission abort is severely restricted. Essentially, there is no free return to a LEO node. Either a wait at

some location or a large plane change with the associated performance penalties is required to get back to a LEO node after a lunar swingby.

Figure 2-1.3-1 shows the relative orientation between the Earth-Moon plane and the SSF (or some LEO operational node in an equivalent orbit). Only those times when the out-of-plane angle is low are launch opportunities possible because of the high ΔV penalty associated with any appreciable plane change.

Even when the SSF line of nodes is properly aligned for a lunar transfer there is a relative angle between the orbit plane and the Earth-Moon plane (Figure 2-1.3-2). A lunar transfer leaving from the SSF orbit will remain essentially in that orbit plane. This has a significant impact on the abort scenarios if they require rendezvous with the Space Station on return to LEO.

During the time that the lunar transfer vehicle leaves Earth, swings around the Moon, and returns to LEO on an Apollo-type free-return trajectory, the Space Station orbit plane has rotated as a result of nodal regression. This regression rate is about 7 degrees per day so after a 6-day roundtrip the nodes of the transfer orbit and the SSF orbit will be misaligned by about 40 degrees. Lunar return trajectories can be rotated in plane about the Earth-Moon line with a modest performance penalty. This effectively changes the return transfer orbit inclination. However, changes in ascending node are very expensive in terms of ΔV. This is depicted in Figure 2-1.3-3.

In the event of a need for earliest possible return to the Space Station, the overriding problem is the potentially large (up to 57 degrees) angle of the Moon out of the plane of the Space Station's orbit. (Nominal mission event times are based on the passages of the Moon through this plane, and the opportunities average about 9 days apart.) Figure 2-1.3-4 reflects this worst case condition in the three upper solid "ΔV required" lines. Even a so called "free return" from translunar trajectory cannot avoid the problem because the Moon is, in general, out of the plane at the time of flyby. The point "B" chosen for the plane change maneuver is a location minimizing ΔV. Any approach azimuth at "A" is available. Note that the data presented in Figure 2-1.3-4 were generated for the 90-day study reference vehicle (2.5-stage, LEV/LTV scenario, using LOR) and is presented here to provide visibility into concerns that must be addressed.

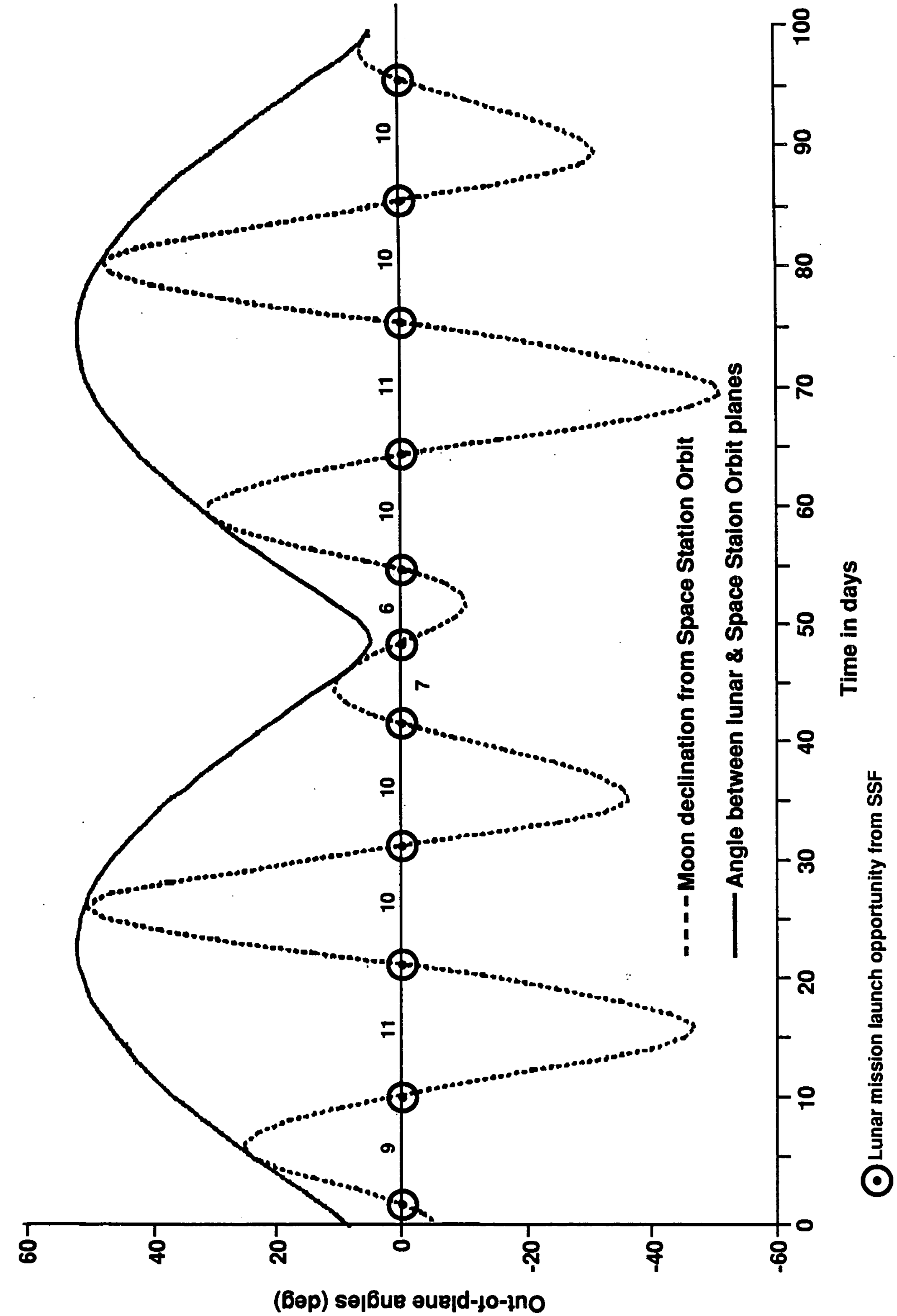


Figure 2-1.3-1. Lunar Launch Windows From SSF

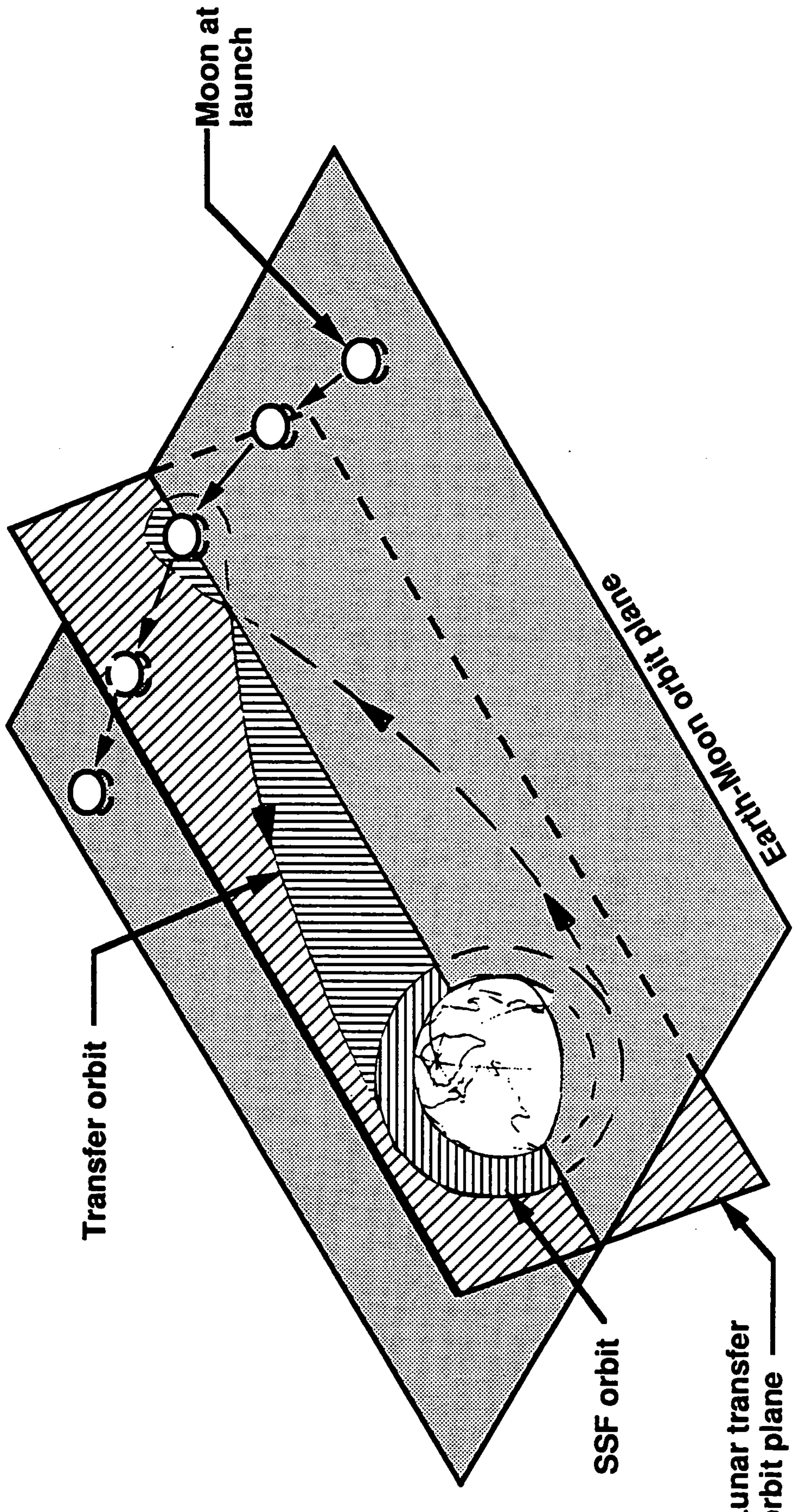


Figure 2-1.3-2. Lunar Transfer Geometry From SSF

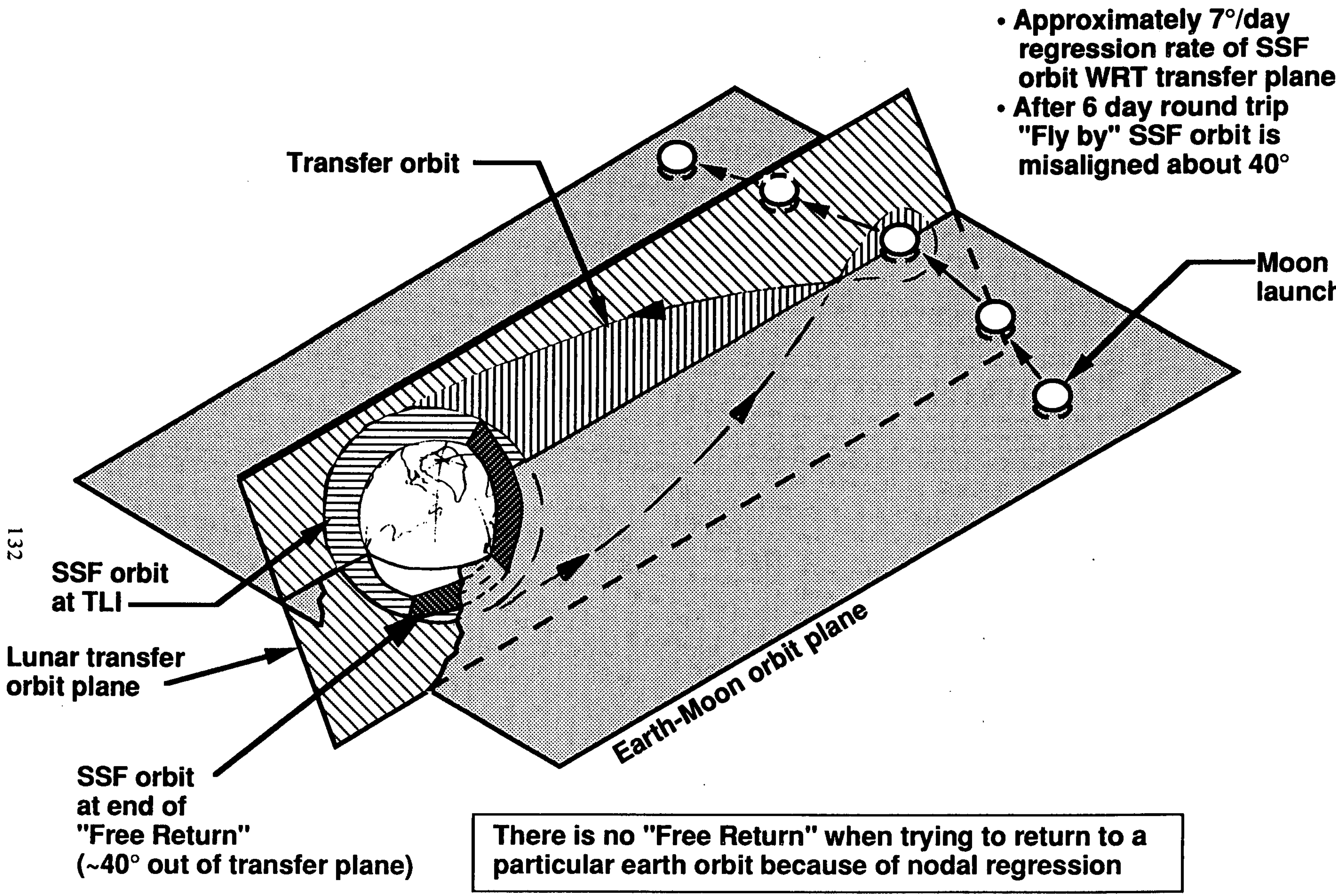


Figure 2-1.3-3. Lunar Transfer Geometry From SSF

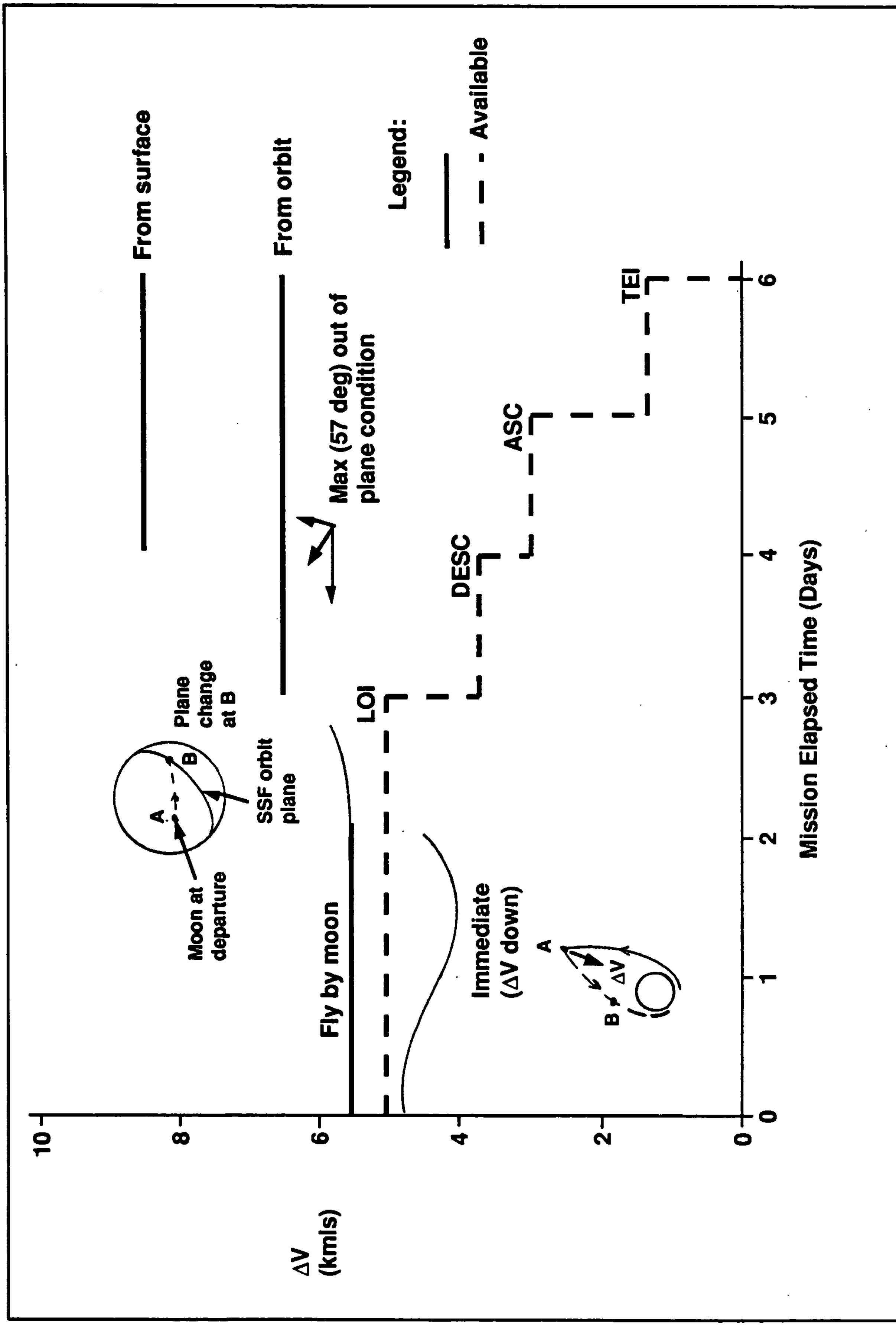


Figure 2-1.3-4. Abort Scenario Capability

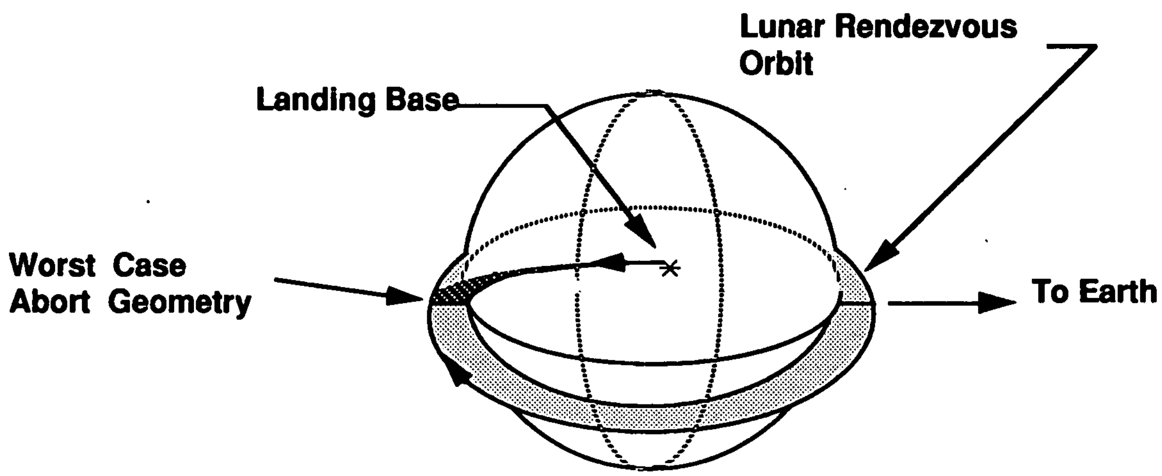
One way around the problem early in the mission, post-TLI burn, is shown as the "immediate" return. Here, a downward ΔV reverses the radial rate. The Space Station orbit thus has less time to regress, though the increasing plane change requirement is seen in the upturn of this line. A nominal mission can be planned that reduces the ΔV requirement by launching when the SSF and lunar alignment favors the in-plane geometry.

Note that these free-return issues are only applicable to STV concepts that use a LEO node. Both ground-based options (GB-1.5S and GO-1.5S) have a free-return capability as the ballistic reentry crew module can return the crew to Earth.

Options for accommodating aborts with a LEO-based concept depend on the mission phase and situation requiring the abort. Options include inclusion of the necessary ΔV capability (large performance penalty), the use of a rescue vehicle to retrieve the crew from a LEO (non-aligned with the LEO node) obtained after an abort return, or waiting until the LEO node orbit is in the necessary alignment either through (1) use of a LEO parking orbit to wait until the parking orbit and the LEO node orbit are aligned, (2) waiting in a LLO orbit, or (3) waiting on the lunar surface (either of these may require a long wait time, which may be undesirable in emergency situations).

The operational scenarios that have a node to rendezvous with in low lunar orbit (LOR approach) were not selected. However, if the LOR approach is ultimately chosen, there are times when, depending on landing site latitude and lunar node orbit inclination, additional ΔV must be available for immediate return; otherwise, safe haven must be available while you wait on the surface for proper alignments. However, the non-optimum lunar orbit operations do not have a severe performance penalty associated with them as do the Earth-orbiting node non-optimum operations.

As the Moon revolves on its axis and rotates around the Earth, the lunar orbiting segment will remain in a fixed inertial attitude. The orbit will not pass over the landing site and in fact can be some distance away depending on the site selection and node orbit. This is depicted in Figure 2-1.3-5 where the landing base and orbit are shown in their worst misalignment. To rendezvous with the



- Considerations for use of LOR
- For inclination = latitude plane change = 2 * i
- For 10 ° orbit and landing site the additional above coplanar ΔV is 468 mps (20 ° plane change)

Figure 2-1.3-5. Abort From Lunar Surface Considerations

orbiting element, an LEV would have to ascend to orbit and then make a plane change to match orbits. The ΔV to perform this plane change is shown for a 10-degree orbit inclination and 10-degree landing site

2-1.4 SEPARATE PILOTED AND CARGO ANALYSIS

2-1.4.1 Introduction

The current STV space-based and ground-based concepts perform both lunar surface piloted and cargo missions with a common vehicle design optimized for 21 piloted missions and 4 cargo-only missions. This analysis addresses the effects of varying the amount of delivered cargo per mission and varying the number of cargo-only missions for two design cases: (1) separate vehicle designs for the piloted and cargo missions (small piloted vehicle and large cargo vehicle) and (2) a common vehicle design for both piloted and cargo-only missions (optimized cargo split). Figure 2-1.4.1-1 gives a summary of the analysis assumptions and analysis design cases for the space-based and ground-based vehicles.

2-1.4.2 Performance Comparison

Space-Based Vehicle Concept. The space-based concept is a direct-to-the-surface vehicle, shown in Figure 2-1.4.2-1, and includes a reusable aerobrake, crew module, six-engine core vehicle that returns to a LEO node after piloted missions, and four sets of droptanks, with two sets expended after the TLI burn and two sets expended after the lunar descent. For the separate design cases 1 through 4, the cargo delivery vehicle includes an expendable core stage and TLI droptanks that are larger than those on the piloted vehicle. For the common-core design cases 5 through 8, the same core vehicle is used for both piloted and cargo-delivery missions and is flown in an expendable mode without an aerobrake or crew module for the cargo-delivery missions. Figure 2-1.4.2-2 shows the relationship between piloted cargo-delivery capability and cargo-only capability for the space-based common-core design case.

Objective:

Determine the effect of varying the cargo split between piloted and cargo lunar missions for the following:

1. Separate piloted and cargo vehicle designs
2. Common core vehicle design

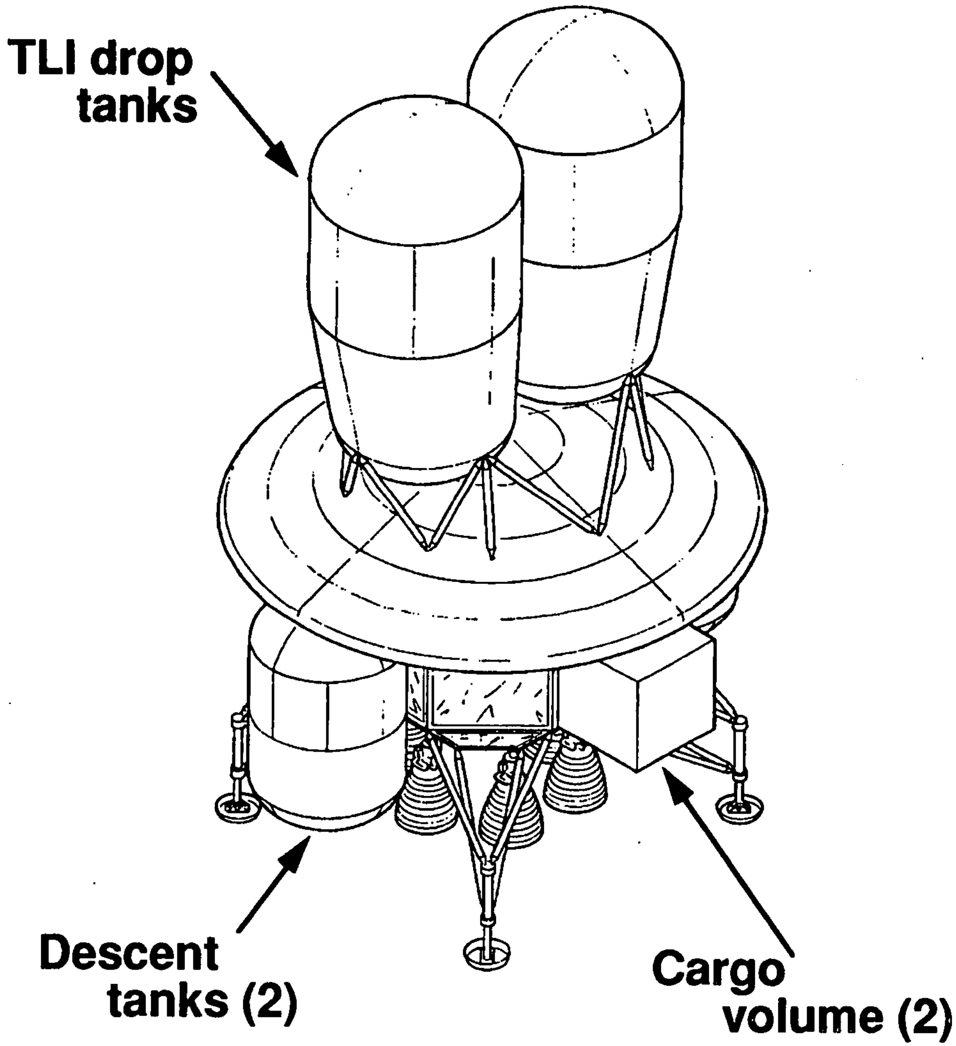
Assumptions:

1. Total number of piloted missions: 16 steady-state, 5 replacement.
2. Total cargo to lunar surface over mission model = 418 t
3. Cargo is 'rubberized' - can be divided in any manner.
4. Optimimum cargo split determined by performance, not structural constraints.

Options:

	No. Missions		Cargo per Mission (t)	
	Cargo	Pilot	Cargo	Pilot
Separate Piloted / Cargo Veh Design				
OPTION 1	4	21	104.5	0.0
OPTION 2	4	21	94.0	2.0
OPTION 3	6	21	62.7	2.0
OPTION 4	8	21	47.0	2.0
Single Core Vehicle Design				
OPTION 1	8	21	45.3	2.7
OPTION 2	6	21	48.7	6.0
OPTION 3	4	21	52.7	9.9
OPTION 4	2	21	57.3	14.5
OPTION 5	0	21	0.0	19.9

Figure 2-1.4.1-1. Separate Piloted and Cargo Analysis Assumptions and Cases



Features

- 50 ft dia aerobrake
- Largest element 65 mt (TLI tankset)
- Reentry L/D >.2
- Asymmetric vehicle (offset crew module)
- Launchable in 30 ft shroud
- 15 ft x ∞ cargo envelope (expendable missions)
- Recovery to SSF
- Reuse all high value elements
- Crew module fits in Shuttle cargo bay
- Self unloadable

Figure 2-1.4.2-1. Space-Based Vehicle

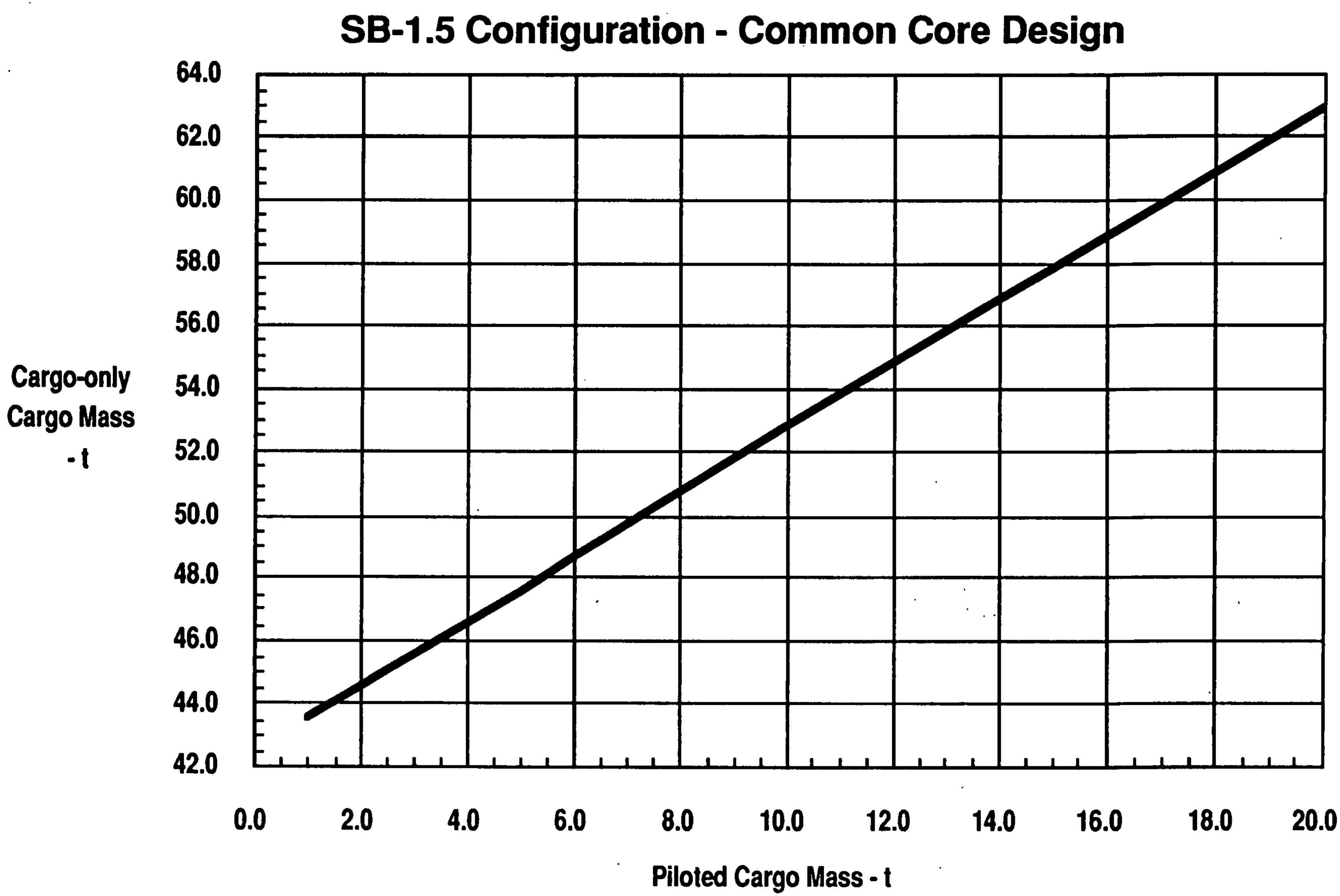


Figure 2-1.4.2-2. Cargo Mass Optimization

Figure 2-1.4.2-3 shows the relationship of Earth-to-orbit (ETO) mass to delivered cargo mass for the space-based piloted vehicle, the space-based common-design cargo-delivery vehicle, and the space-based separate design cargo-delivery vehicle. The current baseline vehicle is indicated, with a piloted delivery capability of 9.9 tons and a cargo delivery capability of 52.7 tons. The total ETO mass for the separate-design cargo vehicle is less than that of the common design because of the more efficient design of the core vehicle. This vehicle does not require separate lunar descent droptanks or the scars for man rating because it is designed only for the expendable cargo-only mode.

A comparison of ETO mass per mission for the different analysis cases is given for the space-based concept in Figures 2-1.4.2-4. For the separate-design case, the ETO mass of the cargo-delivery vehicle is as much as double that of the reference case. The associated piloted vehicle is 18% smaller than the reference concept. For the common-design case, increasing the number of cargo missions from four to eight reduces the ETO mass per mission by 13.3%. Reducing the number of cargo missions from four to zero increases the piloted mission ETO mass by 18%.

A comparison of total ETO mass over all piloted and cargo flights is shown in Figure 2-1.4.2-5. For the separate-design case, the total mass varies from the reference concept by only about 1%. For the common-design case, the total mass varies from the reference by less than 3%.

Ground-Based Vehicle Concept. The ground-based concept shown in Figure 2-1.4.2-6 is also a lunar-direct design and includes a reusable crew module that is ground recovered, a core tank module and core propulsion module with four engines, a lander with two descent stages expended on the lunar surface, and a set of TLI droptanks expended after the TLI burn. Because the core tank module is used for only piloted missions, the separate design cases 1 through 4 have larger droptanks and descent stages for the cargo-delivery missions. For the common-core design cases 5 through 9, the descent stages and droptanks are common for both piloted and cargo-only missions.

Figure 2-1.4.2-7 shows the relationship of ETO mass to delivered cargo mass for the ground-based piloted vehicle and the ground-based cargo-delivery

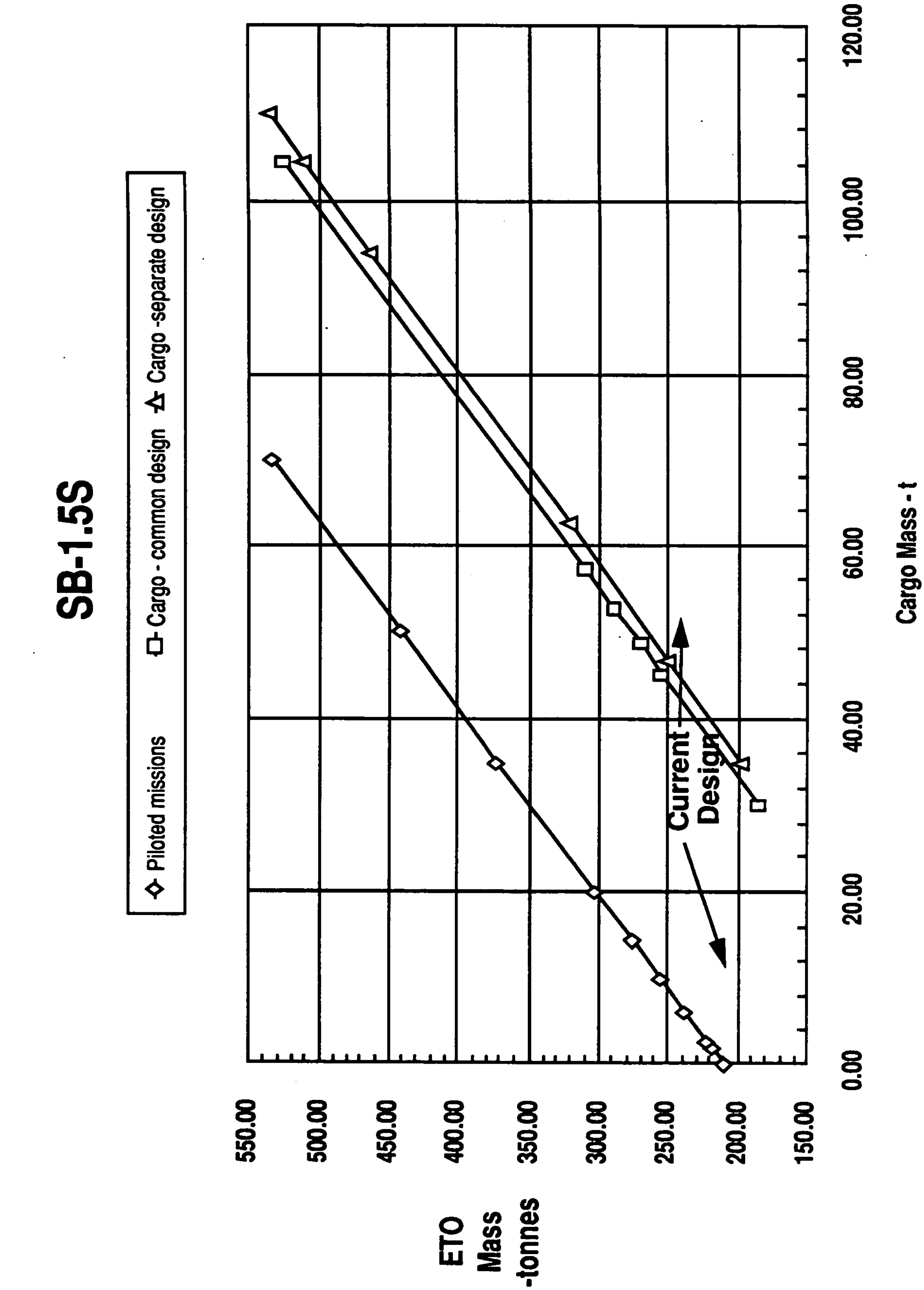


Figure 2-1.4.2-3. ETO Mass Versus Lunar Cargo Mass - SB Concept

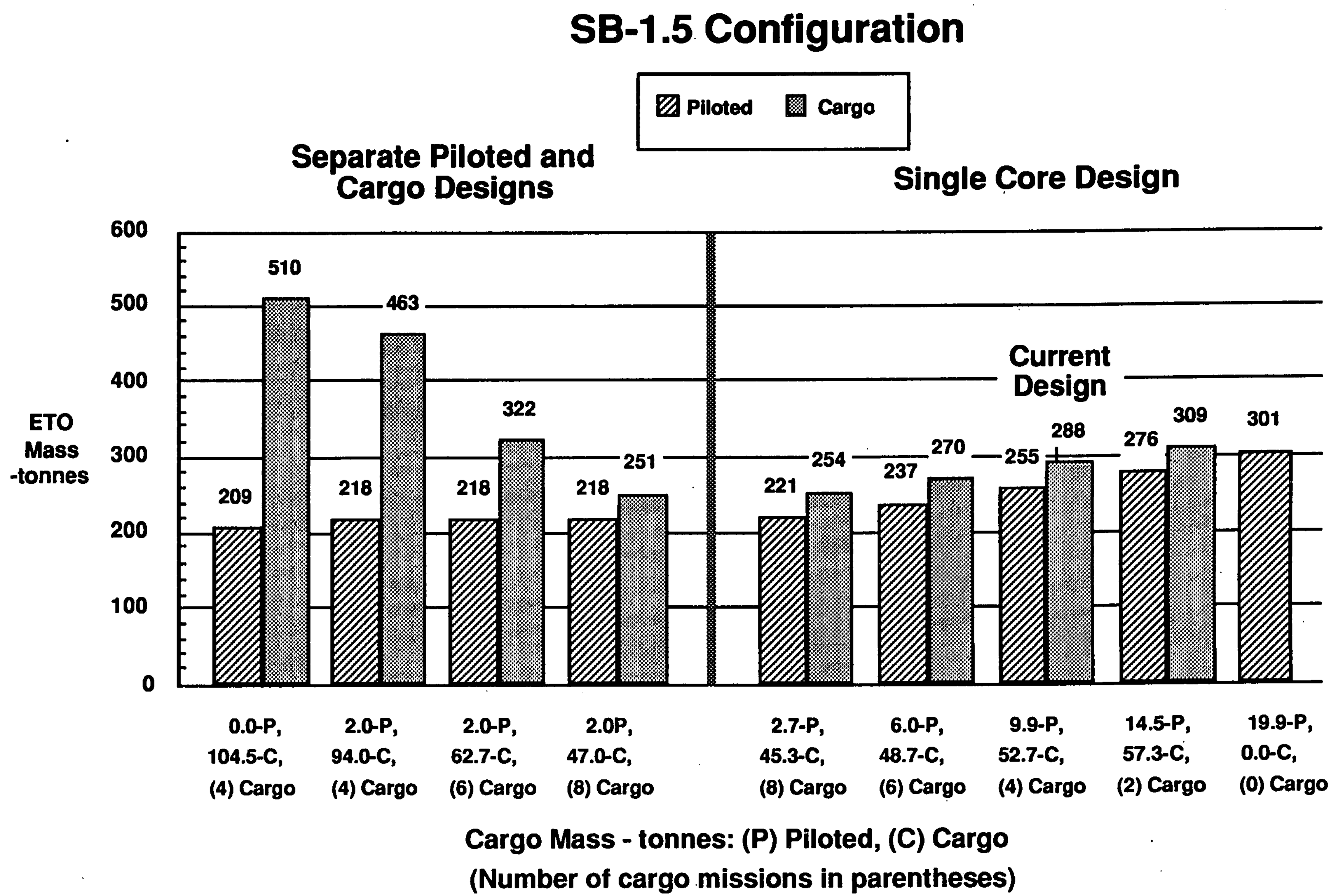
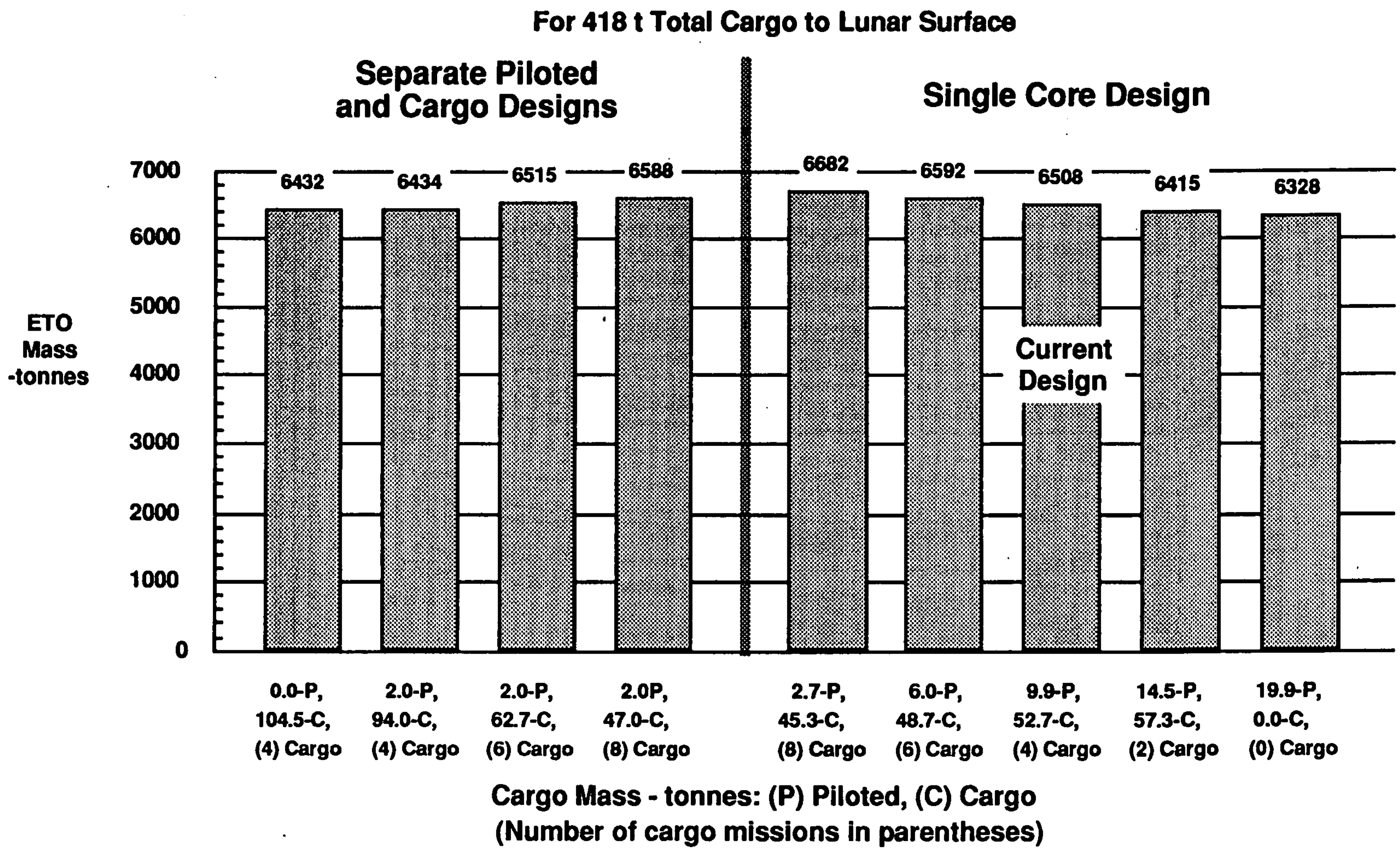
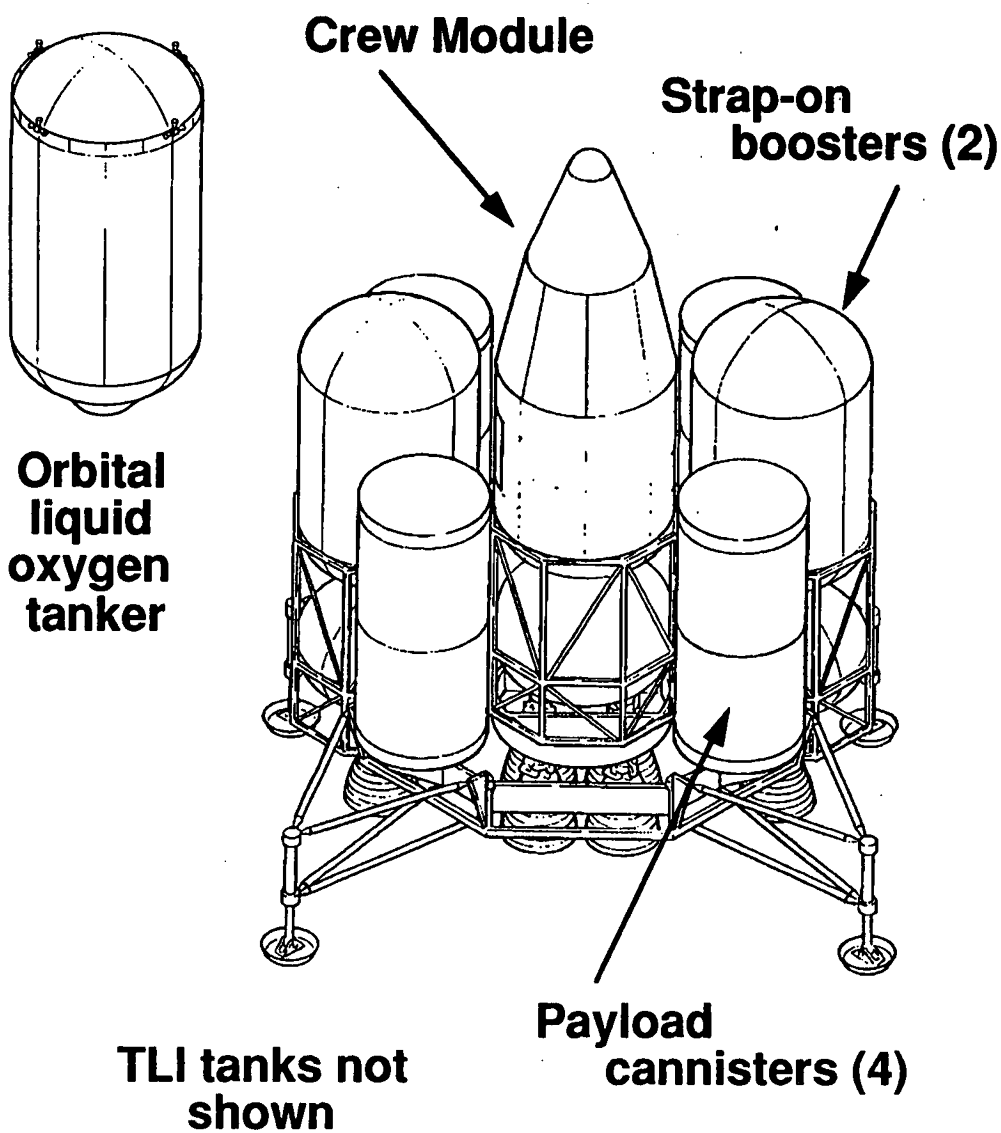


Figure 2-1.4.2-4. ETO Mass Per Mission Comparison - SB Concept



Overall, the minimum number of cargo missions yields the lowest cumulative ETO mass over the mission model.

Figure 2-1.4.2-5. Total ETO Mass Comparison - SB Concept



Features

- No on-orbit vehicle integration
- Tanker launched 1st with oxygen offloaded from stage
- 13.7 m dia launch shroud
- Six 15klb thrust engines
- Cargo envelope: 4.5 m dia x 12.2 m (piloted), 4.5 m dia x ∞ (expendable)
- Self unloadable
- Crew module ground recoverable
- Strap-on boosters usable as separate upper stage

Figure 2-1.4.2-6. Ground-Based Multiple Launch Vehicle

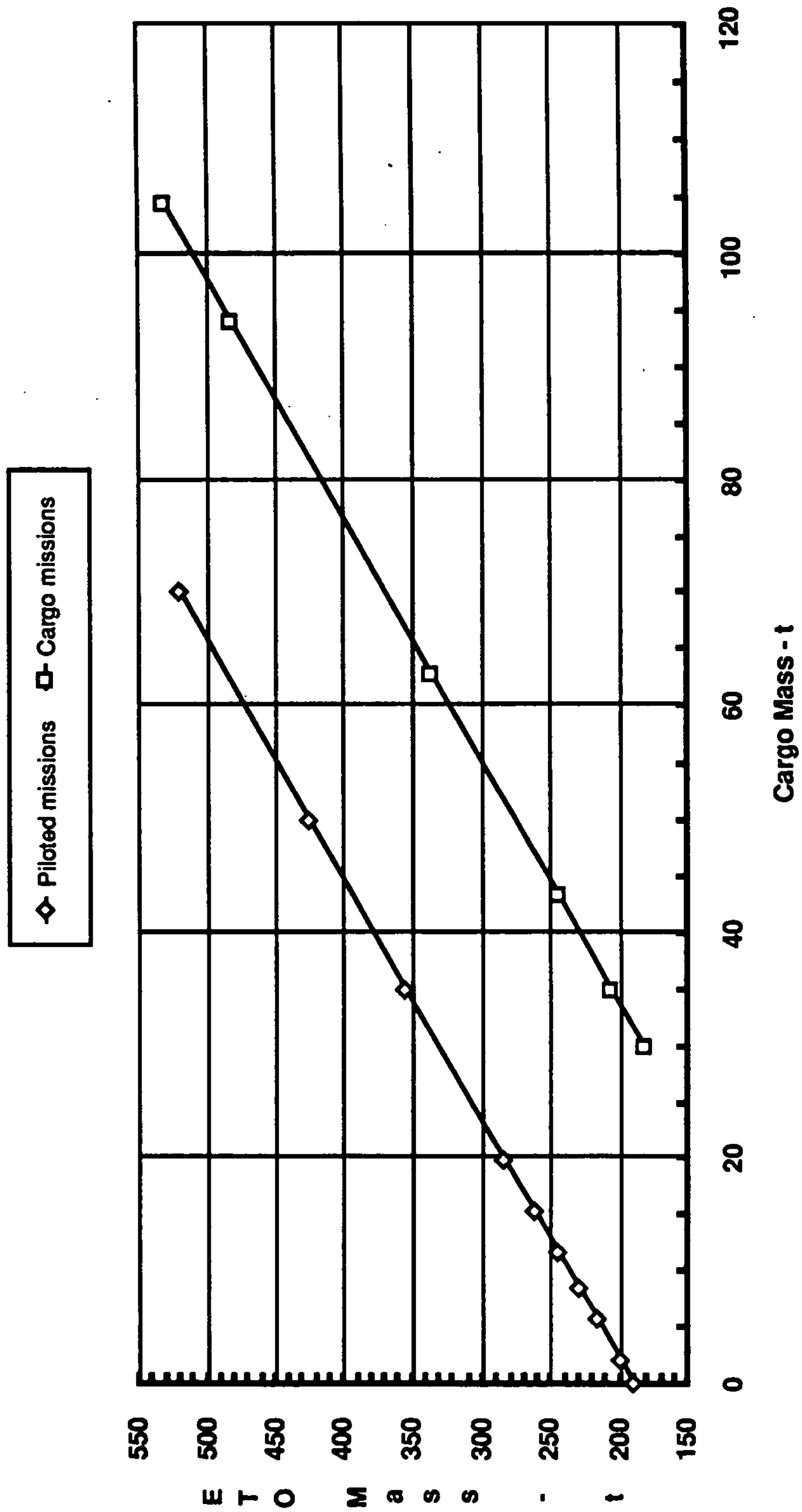


Figure 2-1.4.2-7. ETO Mass Versus Lunar Cargo Mass - GB/GO Concepts

vehicle. The baseline vehicle delivers 11.6 tons of cargo with piloted missions and 43.3 tons of cargo with the cargo-delivery missions.

A comparison of ETO mass per mission for the different design options is given in Figure 2-1.4.2-8. For the separate-design case, the ETO mass of the cargo-delivery vehicle is more than double the size of the reference concept. The associated piloted vehicle is 22% smaller than the reference concept. For the common-design case, increasing the number of cargo missions from four to eight reduces the ETO mass per mission by 11.4%. Reducing the number of cargo missions from four to zero increases the piloted mission ETO mass by 16%.

A comparison of total ETO mass over all piloted and cargo flights is shown in Figure 2-1.4.2-9 for the ground-based concept. For the separate-design case, the total mass varies from the reference concept by less than 3%. For the common-design case, the total mass varies from the reference by less than 3%.

2-1.4.3 Life Cycle Cost Comparison

The final comparison made in this analysis was a LCC analysis, based on the cost model used in the System Architecture Trade Study, presented in section 2-1.1. Launch costs for the analysis were assumed at $100/kg but could be adjusted for current cost estimates without significantly affecting results because of the small differences in total ETO mass in this analysis. Cost results are given in Figures 2-1.4.3-1 and 2-1.4.3-2 for space-based and ground-based concepts, respectively.

For the separate-design case, DDT&E costs were higher because of larger core and droptanks required for cargo missions. The ground-based concept only requires larger droptanks and not a larger core vehicle. For both cases, recurring costs increased as the total number of missions increased and launch costs remained relatively constant due to small variations in total ETO mass. Overall, the minimum cost option has the smallest number of cargo missions.

For the common core design cases, DDT&E showed little variation, but recurring costs showed much larger variations, reflecting the increased number

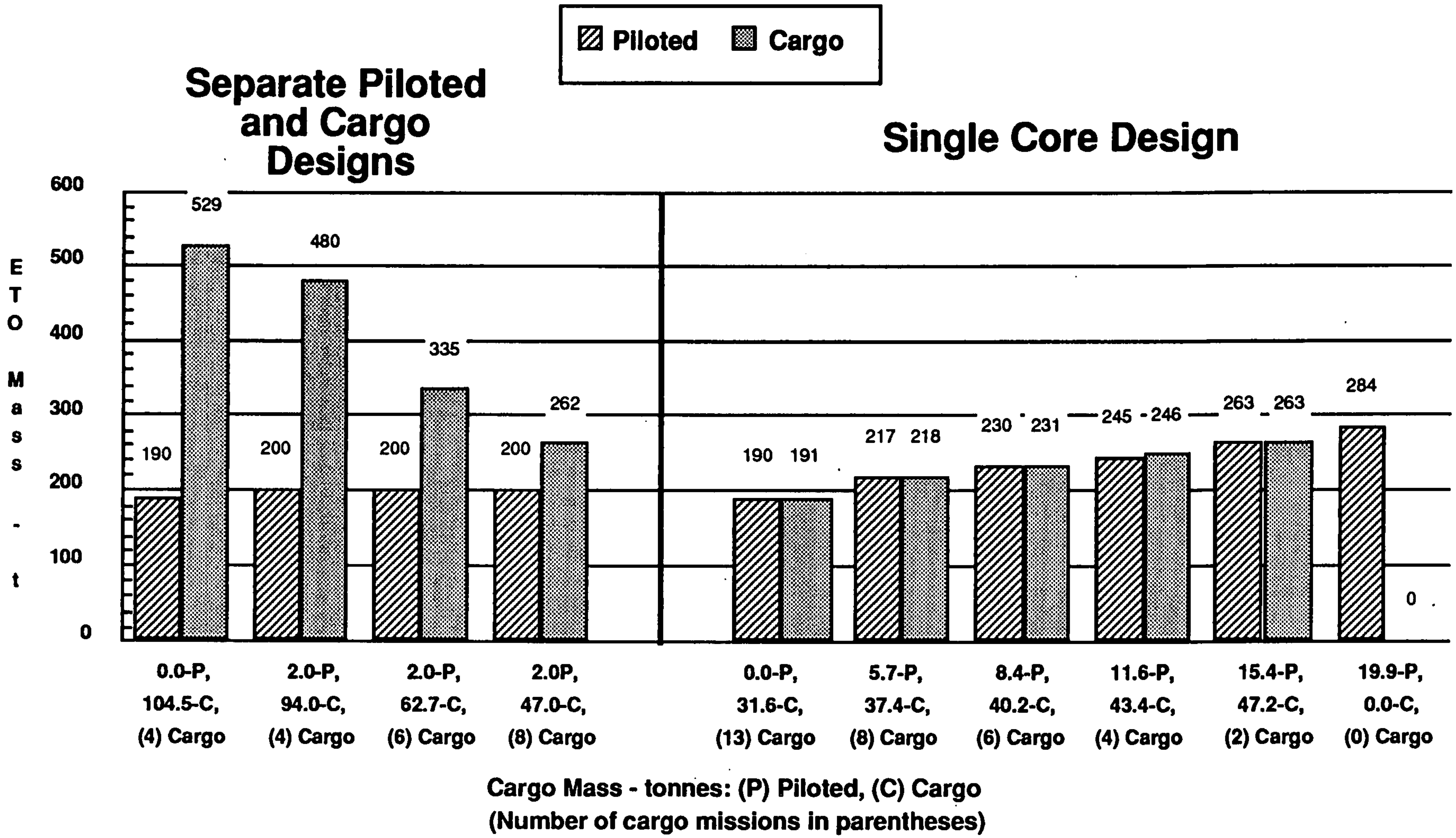


Figure 2-1.4.2-8. Mass Per Mission Comparison - GB/GO Concepts

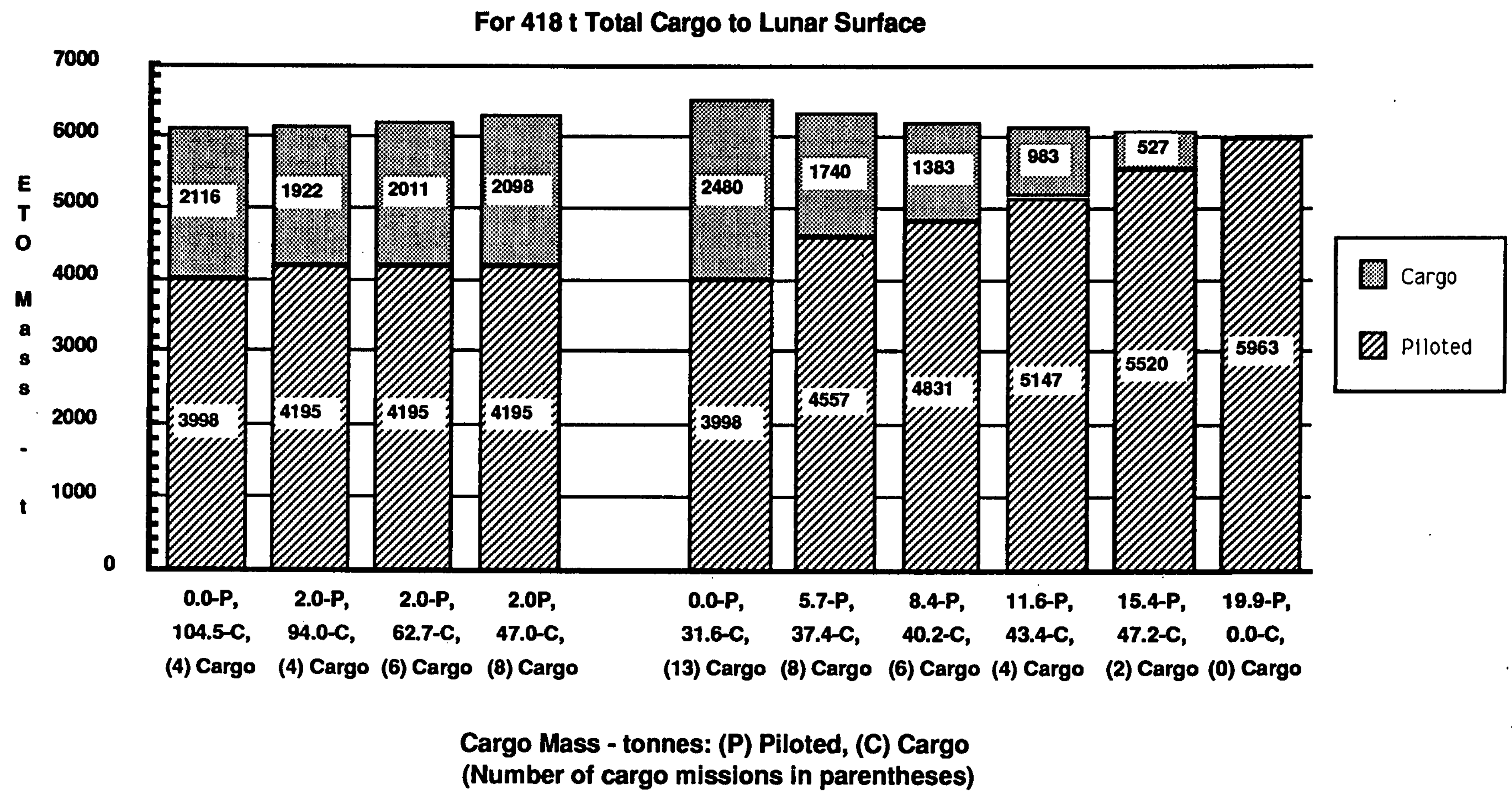
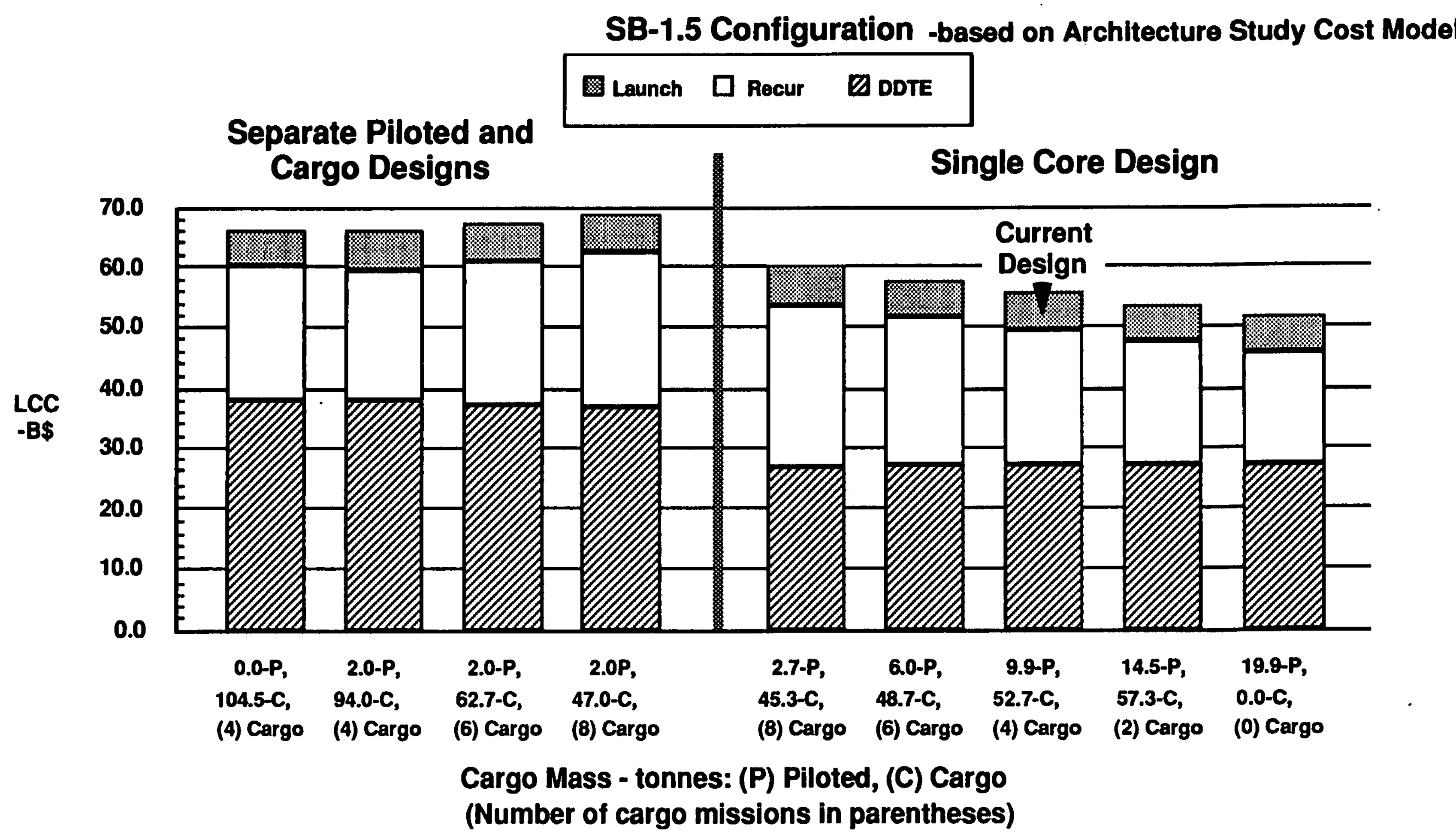
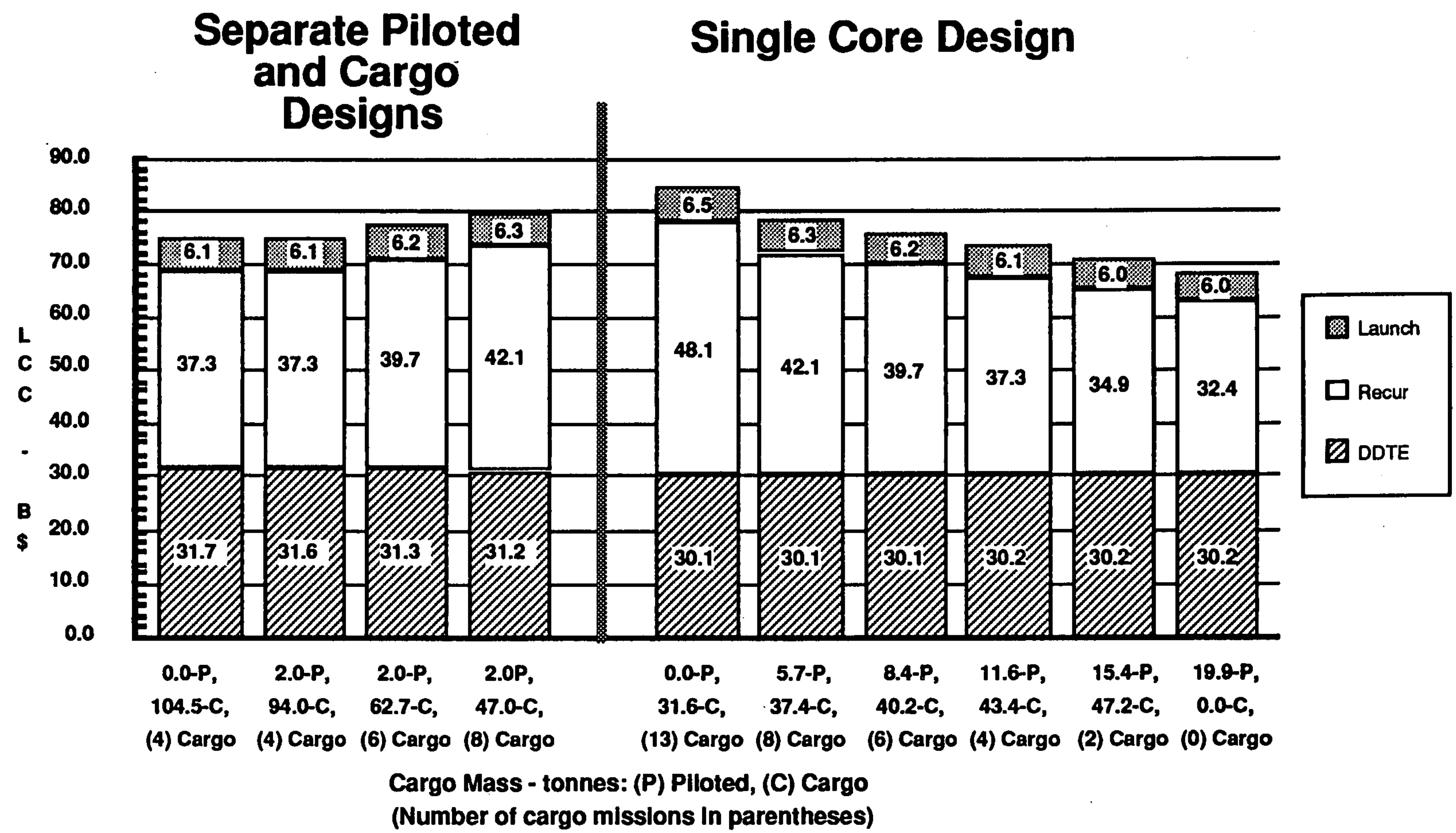


Figure 2-1.4.2-9. Total ETO Mass Comparison - GB/GO Concepts



Single core design has lower cost overall. Lowest cost option is minimum number of cargo missions.

Figure 2-1.4.3-1. Life Cycle Cost Comparison - SB Concept



-based on Architecture Study Cost Model

Single core design has lower cost overall. Lowest cost option is minimum number of cargo missions.

Figure 2-1.4.3-2. Life Cycle Cost Comparison - GB/GO Concepts

of missions. Again, launch cost variations were small. The minimum LCC cases were those with no cargo flights, but had less than 8% variation from the reference and are considered within the error range of the cost model.

In conclusion, although performance per flight favored an increased number of missions with smaller cargo on each mission, both total mission performance and LCC favored the least number of cargo flights. For both space-based and ground-based concepts, the common-design case was favored over the separate-design case in LCC.

2-1.5 STV AS AN UPPER STAGE

2-1.5.1 Introduction

One method of increasing delivered payload capability from an expendable launch vehicle is to deploy the upper stage suborbitally, taking advantage of the typically higher upper stage specific impulse. Historically, upper stages that had high Isp relative to the launch vehicle have been deployed suborbitally to maximize payload capability. Examples include the SIVB stage and Centaur upper stage. Lower Isp upper stages such as the IUS typically have been launched on orbit and do not benefit as much from suborbital deployment.

Because the STV designs were assumed to have high Isp and relatively high thrust, they could benefit from suborbital deployment based on these criteria. At issue, though, is a "go to stay" STV design philosophy demanding high delivered cargo requirements and ready access to space. This analysis addressed these issues and the impact of using the STV as a launch vehicle upper stage.

2-1.5.2 Assumptions and Groundrules

Launch Vehicles. Launch vehicles included in this analysis were the Titan IV with upgraded solid rocket motors (SRM) and an heavy-lift launch vehicle (HLLV) all-liquid concept, such as an Alunar surface. Because partially recoverable Alunar surface designs with propulsion/avionics (P/A) modules are

designed to go to orbit to facilitate P/A module recovery near the launch site, only an expendable HLLV was considered in this analysis.

Because of the differences in the launch vehicle concepts, the analysis differed for the two launch vehicle types. The Titan IV is a fixed design and therefore has launch mass and volume constraints and relatively low engine Isp's (SRM = 286.3 seconds, stage one = 301.5 seconds, and stage two = 316.5 seconds). Therefore, for the Titan IV, the payload delivery capability was maximized by varying STV thrust level and optimizing the Titan IV trajectory to minimize gravity losses.

The HLLV design, on the other hand, is not fixed (launch mass and volume constraints vary according to the chosen design, projected engine Isp is much higher than that of the Titan IV, and projected lift capabilities exceed those of the Titan IV). Therefore, for the expendable HLLV, the gross liftoff weight (GLOW) was minimized by assuming a fixed payload, varying the STV thrust level, and optimizing the launch vehicle trajectory to minimize gravity loss.

STV Design. Important STV design issues addressed in this analysis include the engine throttle ratio between initial thrust and required lunar landing thrust, as well as gravity losses associated with the initial thrust-to-weight ratio. Higher thrust systems minimizing Earth escape gravity losses would require deeper throttling capability for low lunar landing thrust requirements. Conversely, lower thrust systems minimizing depth of throttle requirements would result in much higher gravity loss during Earth escape. Figure 2-1.5.2-1 shows the STV ΔV loss as a function of STV thrust to weight for three design missions, including a GEO delivery mission, an unmanned polar servicing mission, and a lunar cargo delivery mission. The impulsive ΔV shown for the lunar mission is for the TLI burn only. Altitudes, inclinations, and payloads for the three design missions are given in Figure 2-1.5.2-2.

Using an STV design with 90,000-lb initial thrust, 481-second specific impulse, and propellant mass fractions of 0.86 for the lunar lander and 0.87 for orbital delivery stages, the STV cargo-delivery capability from LEO was determined for the three reference missions, as shown in Figure 2-1.5.2-3. Also shown is the typical LEO-delivery capability of the Titan IV and the projected range of Alunar

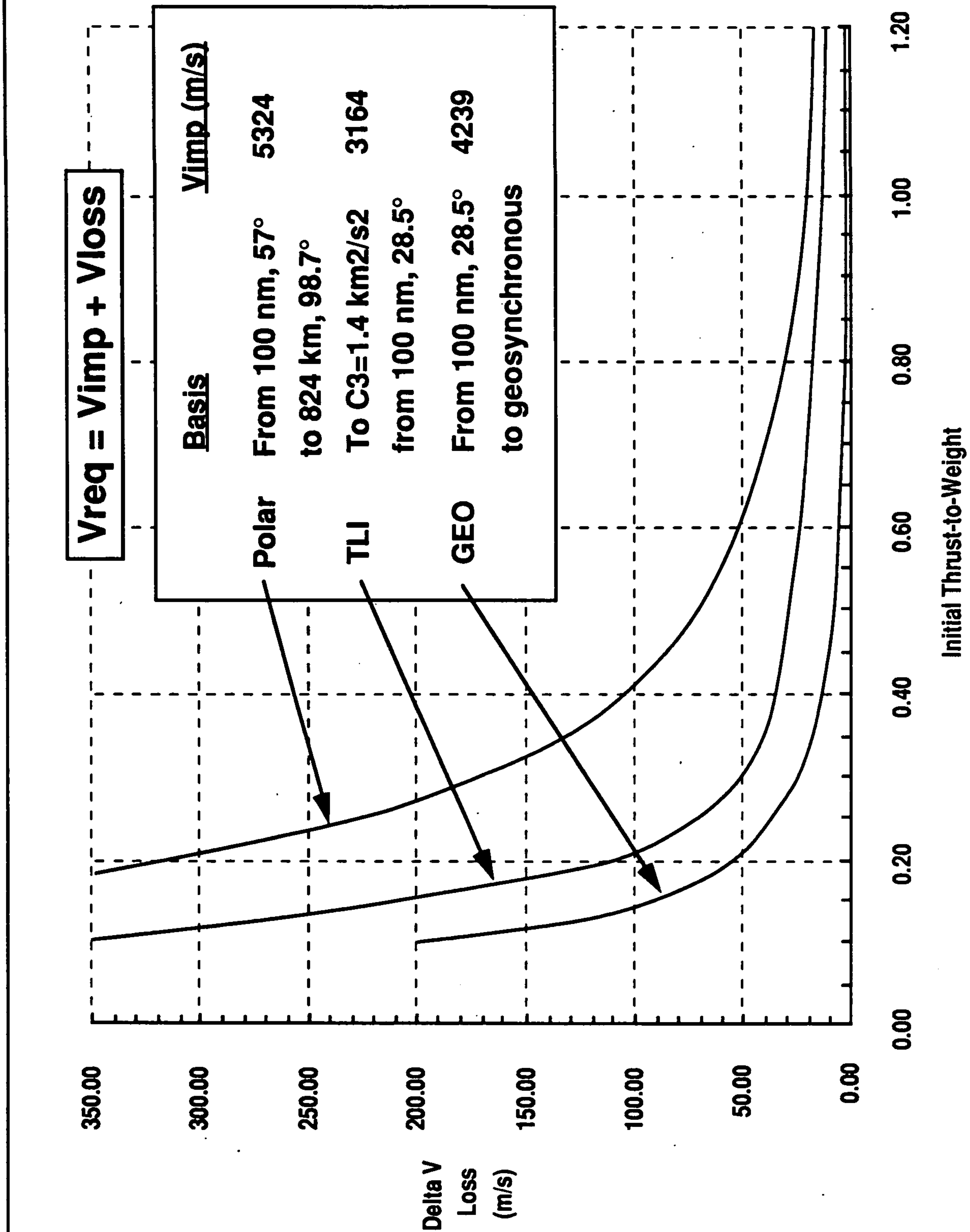


Figure 2-1.5.2-1. Delta-V Loss Comparison

Launch Vehicles:

1.) Titan IV with upgraded SRMs.

- Fixed booster performance - maximize payload delivery capability with variable STV thrust level.

2.) ALS - expendable

- Variable booster performance - minimize gross liftoff weight (GLOW) for a given payload with variable ALS size, variable STV thrust level.

Design Missions:

	Altitude, km	Incl, degree	Payload (kg)
GEO Delivery	35,760 circ.	0°	10,000
Polar Service	824 circ.	98.7°	4500
Expendable Lunar	Lunar direct		34,000

STV: Isp=481 sec, mass fraction (lunar vehicle)= 0.86, mass fraction (other) = 0.87

Figure 2-1.5.2-2. STV as Upper Stage Analysis

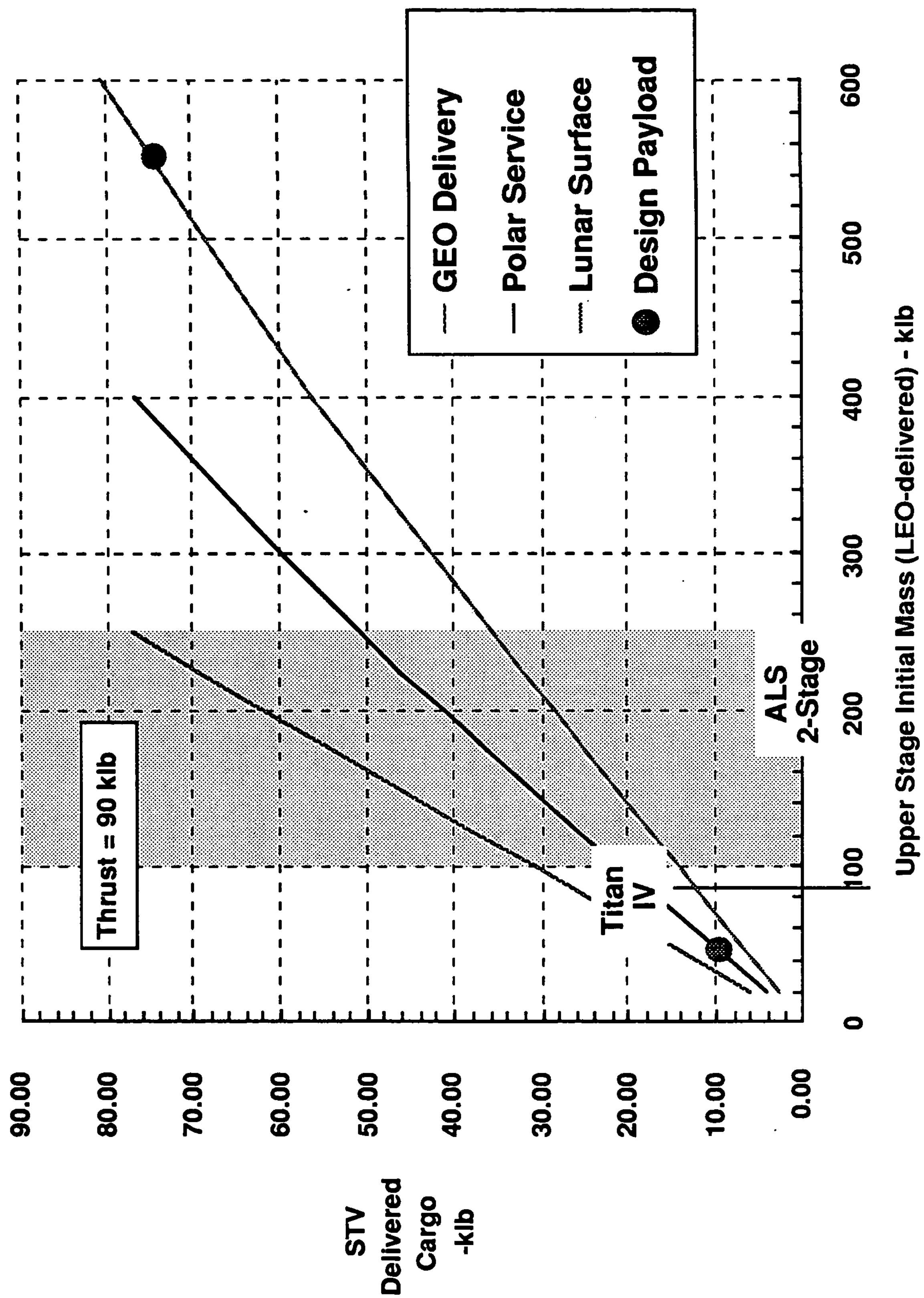


Figure 2-1.5.2-3. STV Delivered Cargo Versus LEO Mass

surface LEO-delivery capability. The polar mission can be flown within Titan IV capability and both the GEO and polar payloads can be delivered within the Alunar surface projected capability. A single-launch lunar cargo mission, however, is far beyond the projected range of both Titan IV and Alunar surface capability.

2-1.5.3 Titan IV Analysis

As was mentioned earlier, the benefit of suborbital STV deployment from the Titan IV was analyzed by maximizing payload delivery capability, with optimum balance between minimum STV throttle range and minimum gravity loss. Figure 2-1.5.3-1 gives the boost profile for an STV suborbital deployment off Titan IV on a translunar injection. The STV stages at an altitude of 265 km and circularizes in LEO prior to the TLI burn.

Upper stage thrust levels at Titan IV separation and at LEO are shown as a function of initial separated mass in Figure 2-1.5.3-2. These thrust levels were obtained by optimizing final orbit mass. A minimum of 20,000-lb total thrust was assumed for initial masses below 80,000 lb. The data are plotted from a minimum mass of 48,000 lb, representing the LEO-delivered capability of the Titan IV.

Figure 2-1.5.3-3 shows upper stage performance as a function of initial separation mass, including burnout mass at LEO and burnout in final orbit for the design missions. Although these data indicates an increase in upper stage delivered performance, the actual delivered payload mass does not necessarily increase, as shown in Figure 2-1.5.3-4. Shown are plots of delivered payload versus initial mass for four destinations, including a lunar free-return orbit, GEO, polar orbit, and the lunar surface. Also shown for the lunar surface delivery are typical throttle ranges between booster separation and lunar landing (at 75% hover thrust).

For the TLI case, the optimum delivered mass was 22,000 lb with a suborbital deployment mass of 80,000 lb. The GEO-delivery case optimized at 15,400 lb delivered with a suborbital deployment mass of 70,000 lb. This payload, however, was only about 200 lb more than that delivered from LEO, with an

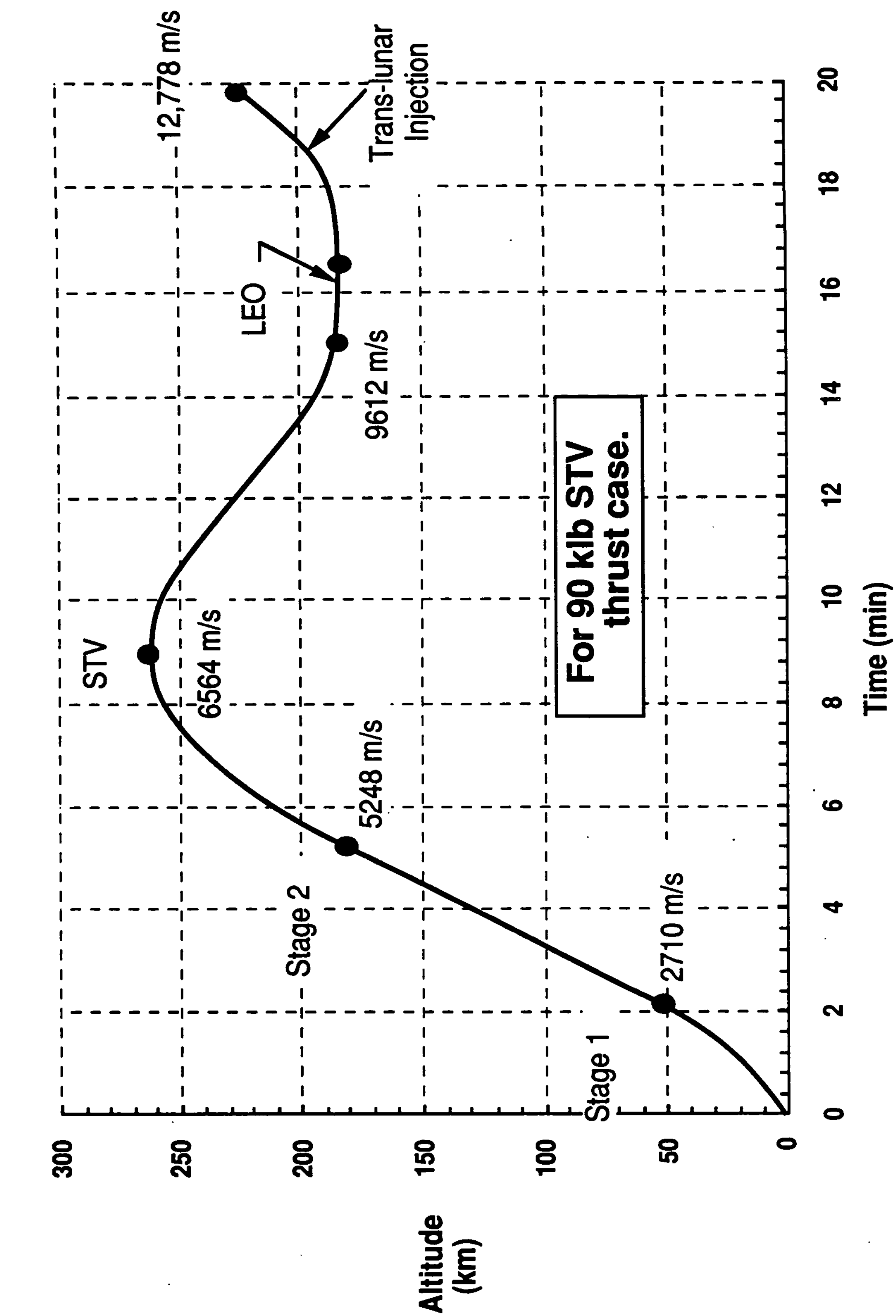


Figure 2-1.5.3-1. Titan IV SRMU Boost Profile

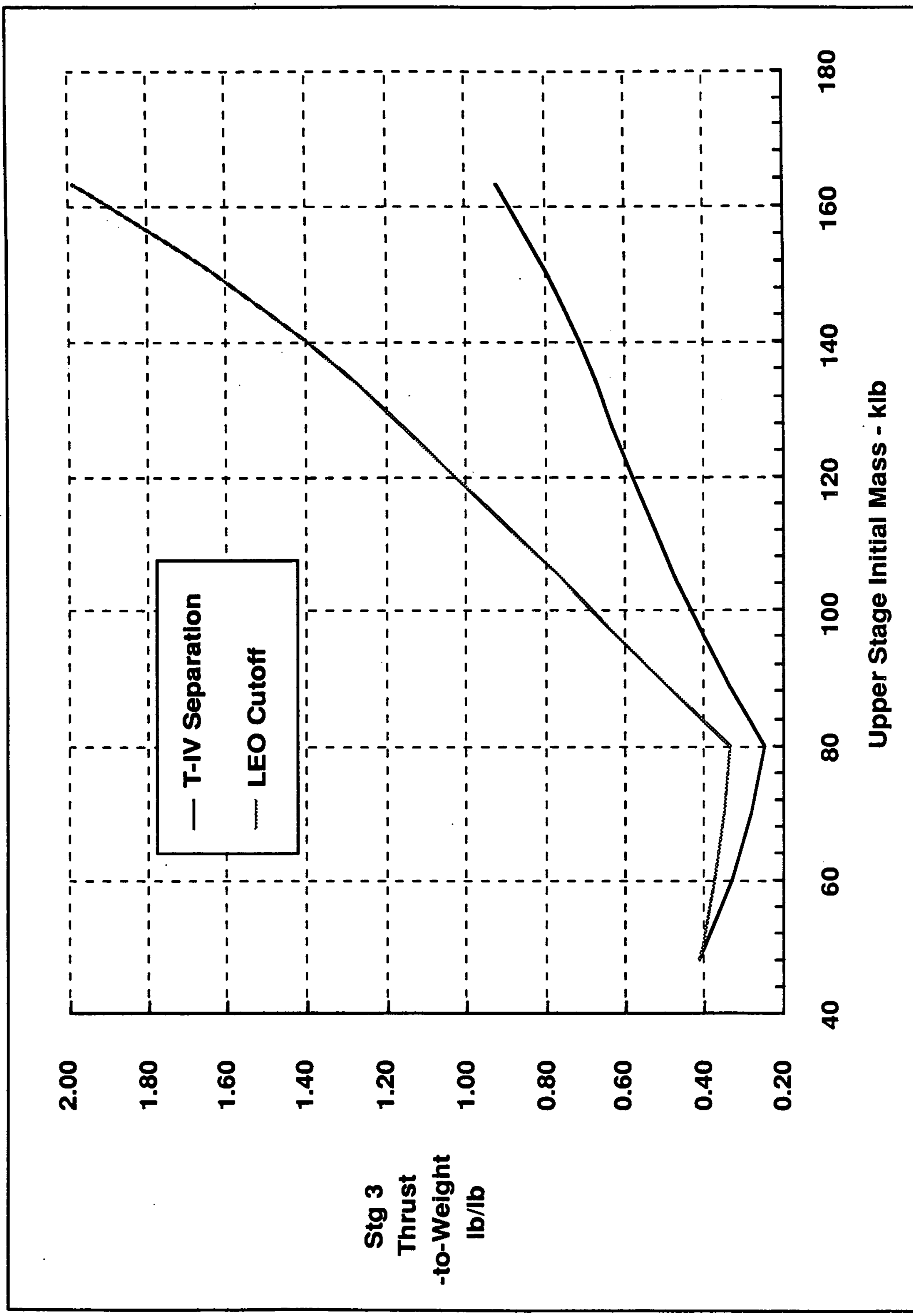


Figure 2-1.5.3-2. Titan IV Stage 3 Thrust and Weight

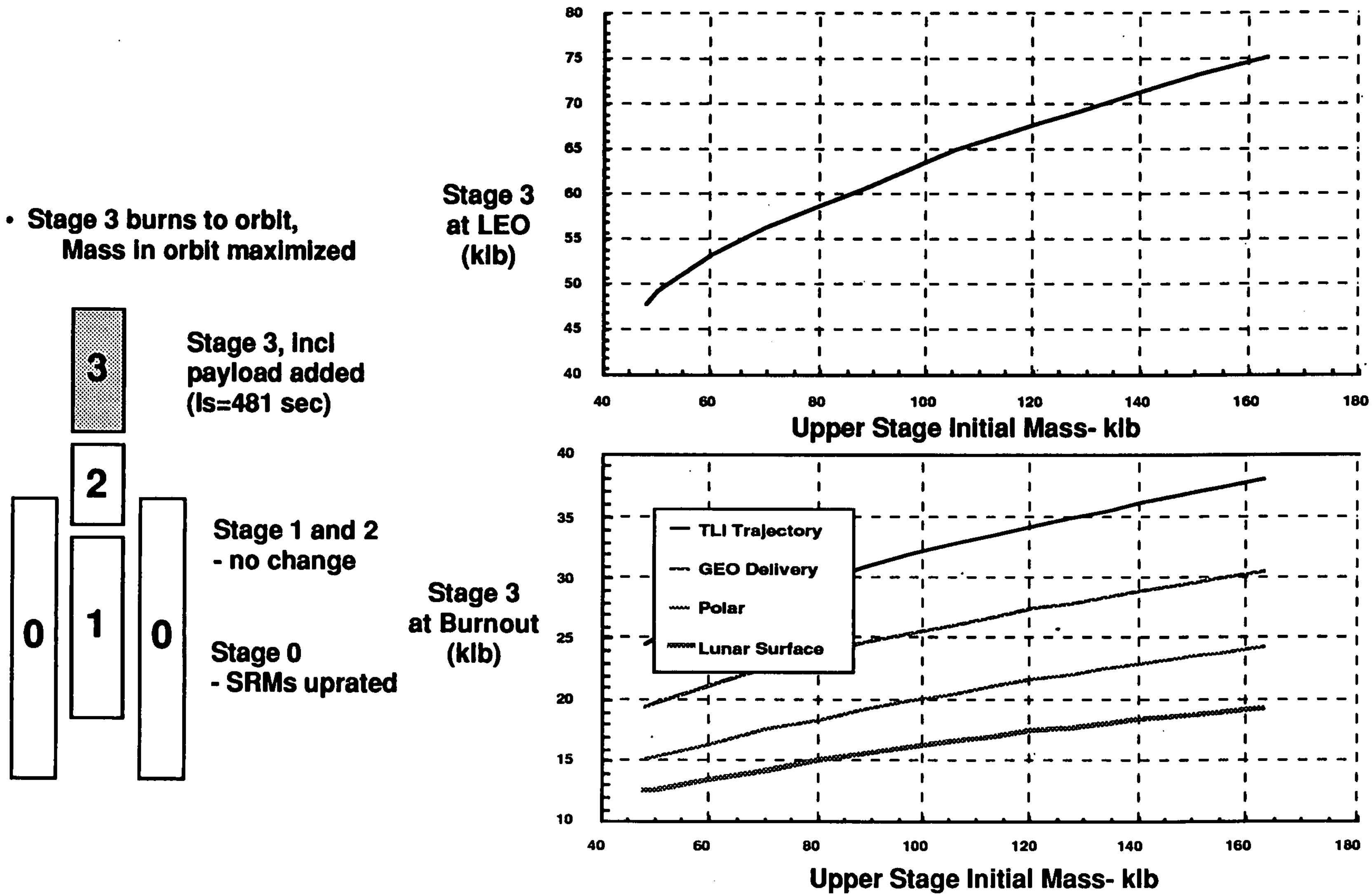
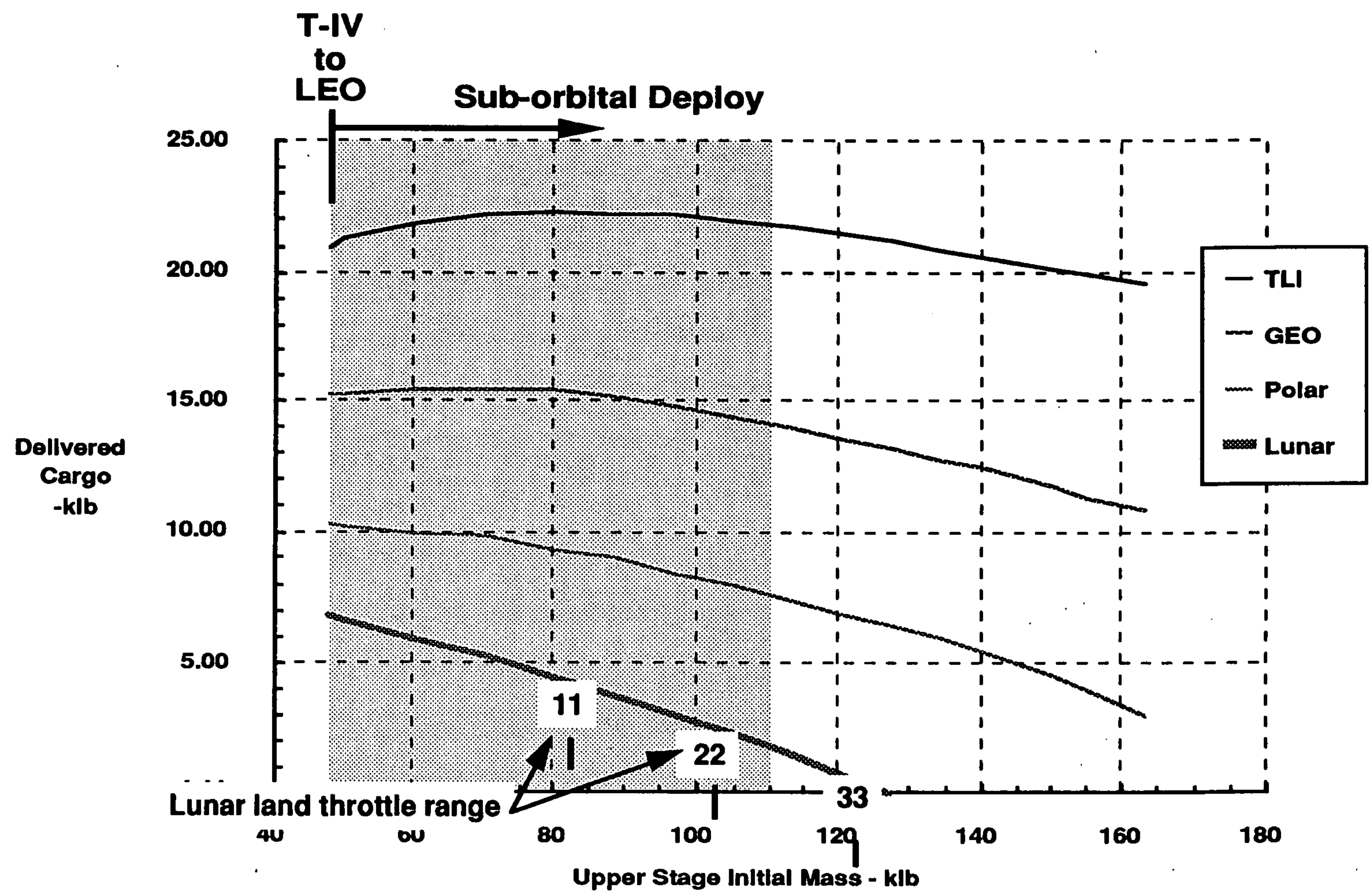


Figure 2-1.5.3-3. Titan IV SRMU and Stage 3 Performance



GEO delivery - Small performance gain with large increase in stage mass.

Polar and Lunar missions emphasize problem of putting too much mass higher up the stack.

Figure 2-1.5.3-4. Titan IV SRMU Stage 3 Delivered Cargo

initial mass of 48,000 lb. Both the polar delivery and lunar surface delivery cases did not optimize suborbitally due to increasing booster losses at higher ΔV requirements.

This analysis emphasized that for a fixed booster capacity, only limited gains can be made by suborbital separation and at the expense of a large increase in upper stage size. Further analysis is needed to determine the benefit, if any, of optimizing delivered payload rather than total delivered mass.

2-1.5.4 HLLV Analysis

The analysis for determining benefit of separating suborbitally from a "rubber" HLLV differed from the fixed capability analysis of the Titan IV. For this analysis, the measure of goodness of suborbital deployment was assumed to be minimum GLOW of the launch vehicle.

Figure 2-1.5.4-1 shows the boost profile for a suborbital deployment of a 90,000-lb thrust STV from an expendable HLLV on a TLI. The STV stages at an altitude of 160 km and circularizes in LEO prior to the TLI burn.

Varying the upper stage delivered mass at LEO results in gross liftoff masses as shown in Figure 2-1.5.4-2. Shown is the performance difference between a partially reusable Alunar surface and an expendable HLLV. Since the partially reusable Alunar surface must go to orbit for propulsion module recovery near the launch site, only the expendable HLLV was assumed for suborbital STV deployment in this analysis.

Increasing the thrust level of the STV and deploying suborbitally from the launch vehicle decreases the GLOW as shown in Figure 2-1.5.4-3 for the polar- and GEO-delivery cases. Delivery of the 75,000-lb lunar DRM payload was beyond the projected lift capability and was not considered in this analysis. Overall, suborbital deployment of the upper stage resulted in a 8% to 10% decrease in launch vehicle GLOW for the polar mission and a 7% to 13% decrease for the GEO mission. Very little improvement was shown beyond a 60,000- to 70,000-lb upper stage thrust level.

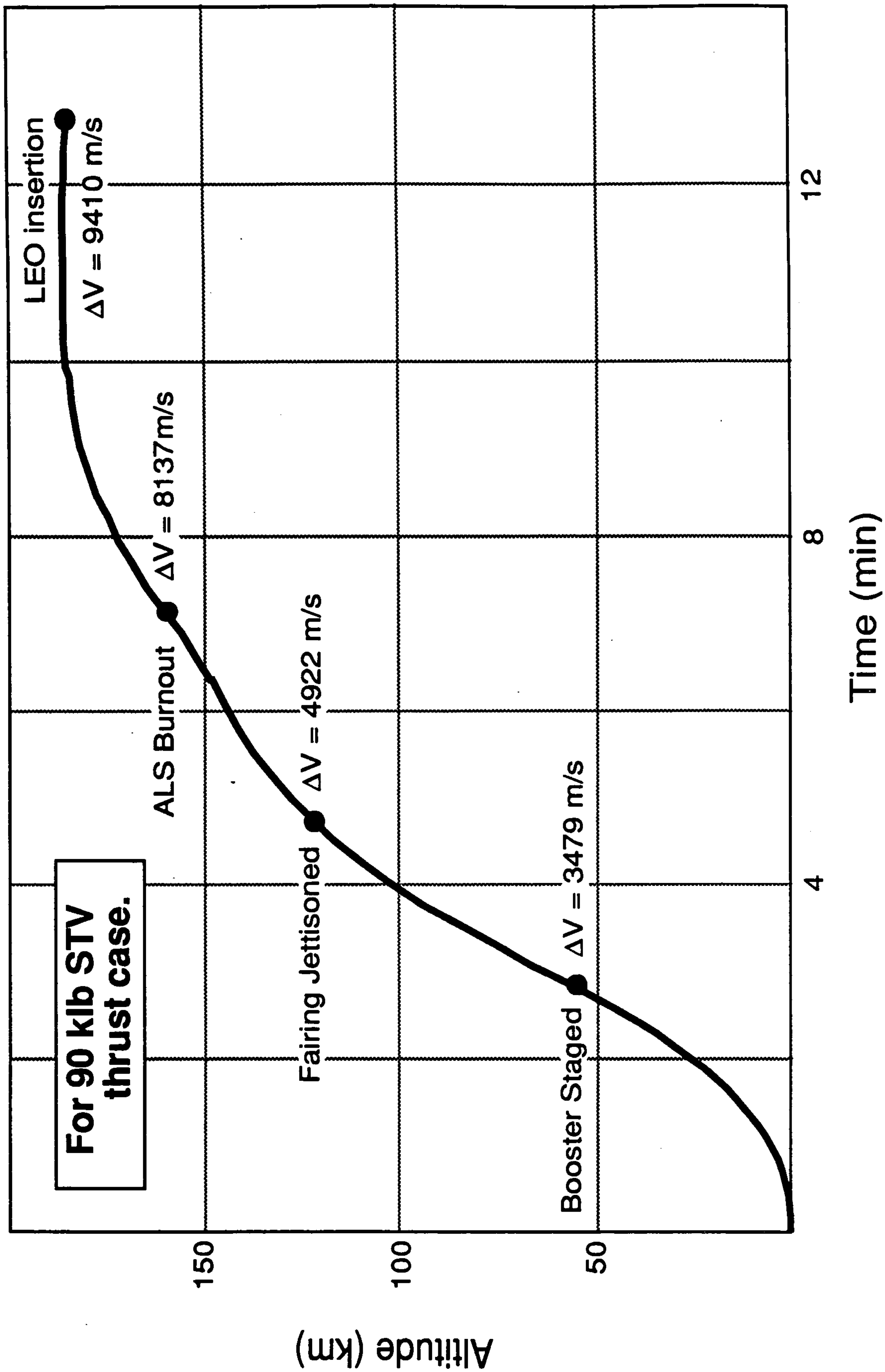


Figure 2-1.5.4-1. Expendable ALS Boost Profile

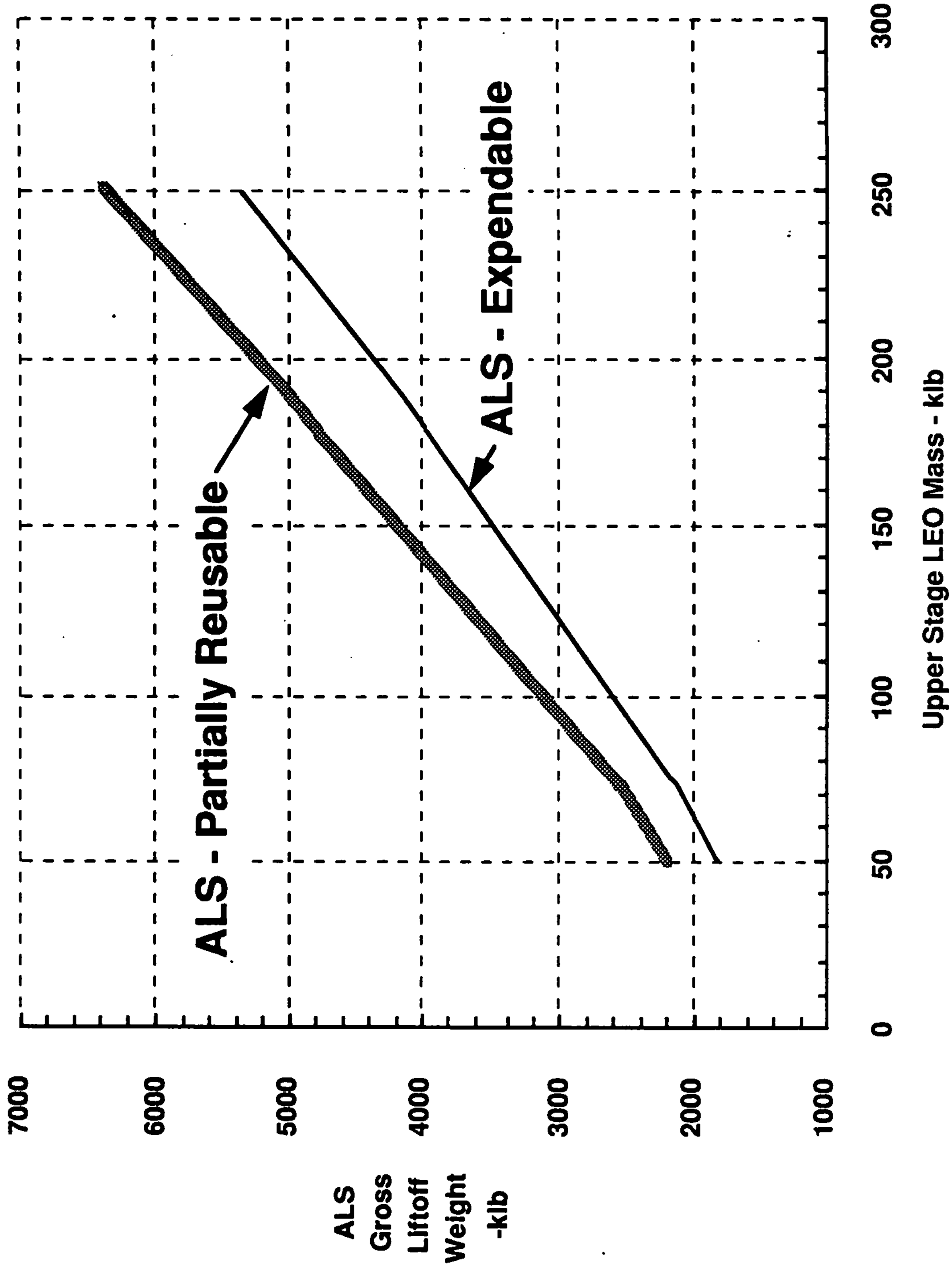
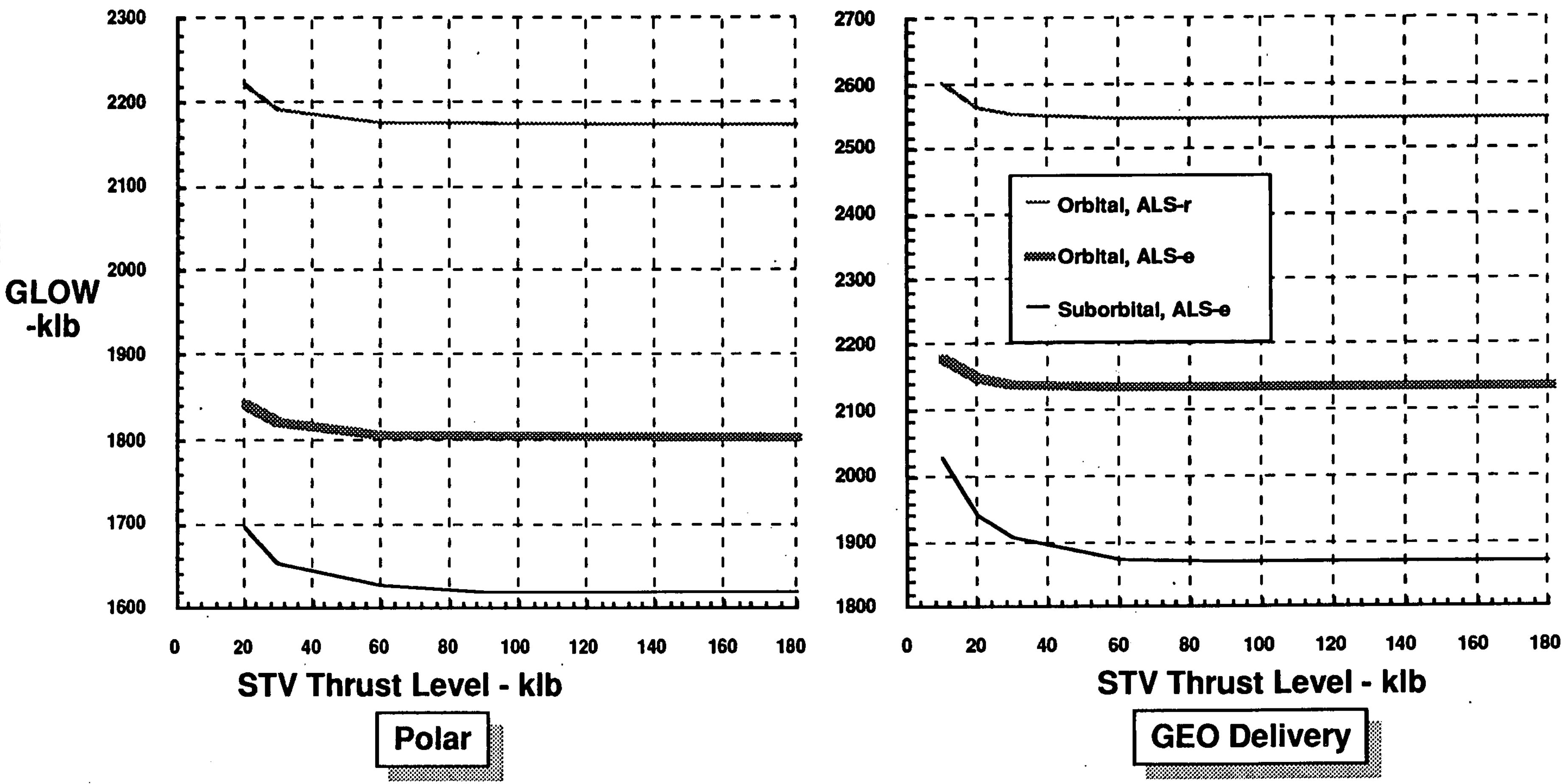


Figure 2-1.5.4-2. ALS GLOW Versus LEO-Delivered Mass



GLOW is minimized with sub-orbital delivery of the upper stage for polar and GEO missions.

Figure 2-1.5.4-3. ALS GLOW Versus STV Thrust

These preliminary results indicated a benefit from deploying suborbitally from an expendable HLLV, but further definition of the launch vehicle capabilities and analysis of a point design need to be accomplished.

2-2.0 PRELIMINARY REQUIREMENTS DOCUMENT

1.0 SCOPE

1.01 Scope

The following document is intended to provide preliminary requirements for a Space Transfer Vehicle with a primary mission to support the transportation requirements of the lunar exploration program. The STV will also be capable of supporting other missions including geosynchronous, planetary, and ultimately evolving to support manned missions to Mars.

The Space Transfer Vehicle Concepts and Requirements Study included investigation of all cryogenic mission architectures including:

1. Space-based and ground-based concepts.
2. Staging options from one to four stages.
3. Lunar orbit rendezvous and direct trajectory options.
4. Single, dual, and hybrid crew module options.

The requirements contained within this document are not intended to preclude any mission architecture option but specific requirements may only apply to a particular architecture.

1.02 Definition

The STV consists of the following four flight elements:

1. Core vehicle (with aerobrake for space based only).
2. Crew modules.
3. Droptanks.
4. Tanker (ground based only).

1.03 Nomenclature

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1.03.01 Requirements Source

The requirements sources are divided into the following three categories:

	Code	Source
1.	(C)	Customer
2.	(D)	Derived
3.	(P)	Provisional

The requirement source code appears beneath the requirement number.

1.03.02 Mission Architecture

The various mission architectures include the following:

Code	Architecture
(Lunar)	Applicable only to lunar mission.
(Piloted)	Applicable only to piloted lunar missions.
(Cargo)	Applicable only to lunar cargo missions.
(SB)	Space based.
(GB)	Ground based.
(LOR)	Lunar orbit rendezvous.
(LOD)	Lunar orbit direct.
(1P/A)	Single propulsion/avionics module.
(LTV/LEV)	Dual propulsion/avionics modules.

If a requirement specifically applies to one of these architectures, the architecture code appears beneath the requirement number.

2.0 APPLICABLE DOCUMENTS

2.01 Government Documents

2.01.01 Mission Model for STV Concepts and Requirements Studies

(C) The mission model supplied by NASA MSFC is based primarily on the civil needs database (CNDB) version FY89. The lunar portion of the CNDB was replaced with the the Option 5 lunar program defined by level 2 at NASA JSC. The CNDB data were also appended with a DoD model supplied by MSFC.

2.01.02 Civil Needs Database Version FY89

(C) The STV mission model includes the CNDB version FY89 models for (1) existing transportation systems, (2) modified planned transportation systems, and (3) assuming planned transportation systems.

2.01.03 Planetary Surface Systems-Reference Mission Option 5

(C) The document, generated by NASA-JSC Planet Surface Systems, describes the Option 5 lunar program with one 6-month lunar mission per year (man-tended capability). The full document title is the "Initial Study Period Results Summary, Planet Surface Systems, Reference Mission - Option 5, Conceptual Design and Development Requirements." This document and a document describing Option 1 with two lunar missions per year were supplied by MSFC in December 1989.

2.01.04 Human Exploration Study Requirements (3/14/90)

(C) The Human Exploration Study Requirements (HESR) document was used as the primary basis of the customer supplied requirements contained in this preliminary requirements document.

2.01.05 Man-Systems Integration Standards (NASA STD-3000)

(C) NASA STD-3000 provides requirement definition for all manned characteristics of the vehicles.

2.01.06 Guidelines for Man-Rating Space Systems (JSC-2321)

(C) Written by the Advanced Programs Office, Systems Definition Branch,Systems Analysis Section at NASA - Johnson Space Center, the document provides supporting requirements for design of the piloted transportation systems.

2.01.07 Manned Spacecraft Criteria and Standards (JSCM 8080)

(C) Design criteria and standards for design and construction of piloted transportation systems.

2.01.08 Space Transportation System Specification

(C) The Space Transportation System specification provides launch service descriptions for the vehicle concepts that use the STS.

2.01.09 Safety Policy and Requirements for Payloads Using the STS (NHB 1700.7a)

(C) The document provides safety requirements for vehicle components integrated on the STS for transport to low Earth orbit.

2.01.10 Implementation Procedures for STS Payloads System Safety Requirements (JSC 3830A)

(C) The document provides safety requirements for vehicle components integrated on the STS for transport to low Earth orbit.

2.01.11 Safety, Reliability, Maintenance, and Quality Provisions for the STS Program

(C) NHB 5300.4 [1d-2])es safety requirements for vehicle components integrated on the STS for transport to low Earth orbit.

2.01.12 Space Station Freedom System Specification

(C)(SB) The Space Station Freedom specification provides interface descriptions for the space-based vehicle.

2.01.13 Space Station Freedom Proximity Operations (JSC 19371)

(C) (SB) Space Station Freedom proximity operations provides procedures and guidelines for operations with Space Station Freedom

2.02 Non-Government Documents

2.02.01 Design Reference Missions (Rev-A 3/22/90)

(D) The DRMs were generated as a part of the STV Concepts and Requirements Study and are based on mission model analysis.

2.02.02 Lunar Transfer Vehicle On-Orbit Processing

(SB) The document provides processing timelines and operations descriptions for servicing vehicles at Space Station Freedom.

3.0 REQUIREMENTS

3.01 Definition

3.01.01 Primary Mission

(C) The STV will provide a cost-effective space-based transportation system capable of supporting a human exploration program resulting in a manned outpost on the Moon.

3.01.01.01 Primary Mission Schedule

(C)(Lunar)

Date	Flight	Mission Type	Reusability
2002	0	Cargo	Expended
2003	1	Cargo	Expended
2004	2	Piloted	Replacement
2005	3	Piloted	2
2006	4	Cargo	Expended
2007	5	Piloted	Replacement
2008	6	Piloted	2
2009	7	Piloted	3
2010	8	Cargo	Expended
2011	9	Piloted	Replacement
2012	10	Piloted	2
2013	11	Piloted	3
2014	12	Piloted	4
2015	13	Piloted	5
2016	4	Piloted	Replacement
2017	15	Piloted	2
2018	16	Piloted	3
2019	17	Piloted	4
2020	18	Piloted	5
2021	19	Piloted	Replacement
2022	20	Piloted	2

2023	21	Piloted	3
2024	22	Piloted	4
2025	23	Piloted	5
2026	24	Piloted	Replacement

3.01.01.02 Primary Mission Manifest

3.01.01.02.01 Total Cargo Mass

(C) (Lunar) The vehicle will be capable of transporting a total cargo manifest on both piloted and cargo/expendable lunar missions of 418.6 metric tons over the first 24 years of the program.

3.01.01.02.02 Crew Size

(C)(Lunar) The system will be capable of transporting four crewmembers from Earth to the lunar surface and back to Earth.

3.01.01.02.03 Space-Based Crew Support Duration (LOR)

(D)(SB&LOR) The space-based vehicle will be capable of nominally supporting the crew for up to 22 days when using LOR (see 3.02.03.01, Safety for Abort Requirements).

3.01.01.02.04 Space-Based Crew Support Duration (LOD)

(D)(SB&LOD) The space-based vehicle will be capable of nominally supporting the crew for up to 12 days when using a LOD trajectory (see 3.02.03.01, Safety for Abort Requirements).

3.01.01.02.05 Ground-Based Crew Support Duration (LOR)

(D)(GB&LOR) The ground-based vehicle will be capable of supporting the crew for up to 20 days when using LOR.

3.01.01.02.06 Ground-Based Crew Support Duration (LOD)

(D)(GB&LOD) The ground-based vehicle will be capable of supporting the crew for up to 10 days when using a LOD trajectory.

3.01.01.02.07 Return Cargo

(C)(Lunar) The vehicle will be capable of returning 500 kg of cargo from the lunar surface.

3.01.01.03 Primary Mission ΔV's

(Lunar)

3.01.01.03.01 Lunar Orbit Rendezvous

(C)(LOR) The STV will be capable of providing the following ΔV's for missions using LOR:

Maneuver	ΔV, m/s
Pre-injection maneuvers	10
Translunar injection	3,300
Lunar transit TCMs	10
Lunar orbit insertion	1,100
Lunar descent	2,000
Lunar ascent	1,900
Lunar orbit operations	50
Trans-Earth injection	1,100
Earth transit TCMs	10
Earth orbit insertion	3,300 (40 m/s for aerobrake)
Earth orbit operations	275

3.01.01.03.02 Lunar Orbit Direct

(C)(LOD).The STV will be capable of providing the following ΔV's for missions using lunar orbit direct trajectories:

Maneuver	ΔV, m/s
Pre-injection maneuvers	10
Translunar injection	3,300
Lunar transit TCMs	10
Lunar descent	2,510
Lunar ascent	2,510
Earth transit TCMs	10
Earth orbit insertion	3,300 (40 m/s for aerobrake)
Earth orbit operations	275

3.01.01.04 Primary Mission Operational Phases

(D)(Lunar)

1. Launch and delivery.
2. Low Earth orbit .
3. Translunar injection.
4. Lunar transit.
5. Lunar orbit insertion (LOR).
6. Low lunar orbit (LOR).
7. Lunar descent and landing.
8. Lunar surface.
9. Lunar ascent.
10. Low lunar orbit (LOR).
11. Trans-Earth injection (LOR).
12. Earth transit.
13. Earth orbit insertion (SB).
14. Low Earth orbit (SB).
15. Descent to the Earth surface (GB).

3.01.02 Evolutionary Missions

(D) The STV system will provide an evolvable transportation system capable of supporting the following missions:

STV Evolutionary Missions	Code
Unmanned planetary delivery	P1
Unmanned geosynchronous delivery	G1
Unmanned molniya delivery	D1
Manned geosynchronous servicing	G2
Unmanned LEO polar servicing	S1
Unmanned LEO space tug	T1
Unmanned nuclear/debris disposal	N1
Manned comet Sample capsule recovery	C1

The STV system will also provide an evolvable transportation system capable of supporting a human exploration program resulting in a manned outpost on Mars (Code M1).

3.01.02.01 Evolutionary Mission Schedule

(D) The STV system will be capable of supporting evolutionary missions with the following schedule:

Mission	First Flight	Flight Frequency (Flights/Year)
Unmanned planetary delivery	1999	1
Unmanned geosynchronous delivery	2005	5
Unmanned molniya delivery	2000	0.5
Manned geosynchronous servicing	2006	1
Unmanned LEO polar servicing	2001	0.5
Unmanned LEO space tug	1999	6
Unmanned nuclear/debris disposal	2010	One flight total
Manned comet sample capsule recovery	2002	1
Manned Mars	2015	0.33

3.01.02.02 Evolutionary Mission ΔV's

(D) The STV will be capable of providing the following ΔV's for the evolutionary missions:

Mission Total	ΔV, m/s
Unmanned planetary delivery	4,451
Unmanned geosynchronous delivery	4,207
Unmanned molniya delivery	4,499
Manned geosynchronous servicing	6,064
Unmanned LEO polar servicing	9,784
Unmanned LEO space tug	38
Unmanned nuclear/debris disposal	4,175
Manned comet sample capsule recovery	2,736

3.01.02.03 Evolutionary Mission Payloads

(D). The STV will be capable of supporting the following payload masses for the evolutionary missions:

Mission	Payload Mass (metric tons)
Unmanned planetary delivery	16.0
Unmanned geosynchronous delivery	10.0
Unmanned molniya delivery	6.8
Manned geosynchronous servicing	4.0
Unmanned LEO polar servicing	4.5
Unmanned LEO space tug	71.0
Unmanned nuclear/debris disposal	25.0
Manned comet sample capsule recovery	0.5

3.01.03 Performance

3.01.04 System Operations

3.01.04.01 Mission Operations

3.01.04.01.01 Extra Vehicular Activity

(C)(Piloted) A minimum of two crewmembers will perform each scheduled EVA and the vehicle will have the capability to support EVA for each crewmember.

3.01.04 Earth-Moon System

3.01.04.01.04 Payload Unloading

(D)(Cargo) The vehicle will be capable of unloading cargo to the lunar surface on the first cargo-expendable mission.

3.01.04.01.05 Space-Based Recovery

(C)(SB) The vehicle will be capable of returning the crew to Space Station Freedom.

3.01.04.01.06 Ground-Based Recovery

(D)(GB) The vehicle will be capable of returning the crew to the surface of the Earth with a controlled dry landing.

3.01.04.01.07 Crew Visibility During Landing

(D)(Piloted) The vehicle will provide crew visibility of two landing pads and the horizon during lunar landing.

3.01.04.02 Ground Operations

3.01.05 Maintenance Concept

3.02 Characteristics

3.02.01 Performance Characteristics

3.02.01.01 Service Life (Reusability)

(C)(SB) Reusable elements of the vehicle will be capable of supporting five flights.

3.02.01.02 Flight Performance Reserves

(P)

1. Main Propulsion - 2% FPR on each ΔV maneuver.
2. Reaction Control - 10% FPR of mission nominal propellant.
3. Electrical Power - 20% FPR of mission nominal reactants.

3.02.01.03 Aerobrake Reentry Velocity

(C)(SB) The aerobrake system will be capable of supporting an aeroassist maneuver with an entry velocity of 11.1 km/s (upper limit of the fast lunar return options).

3.02.01.04 Cargo Capability - LTV/LEV Architecture

3.02.01.04.01 Piloted Steady-State Cargo Capability

(C)(LTV/LEV) The vehicle will be capable of transporting 13.0 metric tons of cargo to the lunar surface on steady-state missions using an LTV and LEV on which an operational LEV has been left in lunar orbit by the previous mission.

3.02.01.04.02 Piloted Replacement Cargo Capability

(C)(LTV/LEV) The vehicle will be capable of transporting a lunar cargo consistent with the vehicle sizing for the piloted steady-state cargo capability on replacement missions using an LTV and LEV in which a new LEV is delivered to low lunar orbit.

3.02.01.04.03 Expendable Cargo Mission Capability

(C)(LTV/LEV) The vehicle will be capable of transporting a lunar cargo consistent with the vehicle sizing for the piloted steady-state cargo capability on cargo-expendable missions using an LTV and LEV.

3.02.01.05 Cargo Capability - Ground-Based Single P/A Module

D)(GB&1P/A)

3.02.01.05.01 Piloted Cargo Capability

(D)(GB&1P/A) The single P/A module ground-based vehicle will be capable of transporting 11.6 metric tons of cargo to the lunar surface.

3.02.01.05.02 Expendable Cargo Mission Capability

(D)(GB&1P/A) The single P/A module ground-based vehicle will be capable of transporting a lunar cargo consistent with the vehicle sizing for the capability to the lunar surface.

3.02.01.06 Cargo Capability - Space-Based Single P/A Module

(D)(SB&1P/A)

3.02.01.06.01 Piloted Cargo Capability

(D)(SB&1P/A) The single P/A module space-based vehicle will be capable of transporting 9.9 metric tons of cargo to the lunar surface.

3.02.01.06.02 Expendable Cargo Mission Capability

(D)(SB&1P/A) The single P/A module space-based vehicle will be capable of transporting a lunar cargo consistent with the vehicle sizing for the piloted mission cargo capability to the lunar surface.

3.02.01.07 Lunar Surface Life Support

(C)(Piloted) The vehicle will be capable of providing crew life support for a minimum of 48 hours after landing on the lunar surface.

3.02.01.08 Propellant Boiloff

(C)(Piloted) The vehicle will be capable of maintaining a propellant boiloff rate ≤ 4% per month during all mission phases.

3.02.02 Physical Characteristics

3.02.03 Product Assurance

3.02.03.01 Safety

3.02.03.01.01 Free-Return Trajectories

(C)(GB) The mission design will be capable of supporting return to Earth on free-return trajectories in the event of an abort during transfer to the Moon.

3.02.03.01.02 Safe Haven

(C)(Lunar) The vehicle will use a safe haven capability at the lunar base in the event of a transportation system failure on the lunar surface.

3.02.03.01.03 Ingress/Egress

(D)(Piloted) The vehicle will provide two means of ingress and egress at all times.

3.02.03.01.04 Cargo Jettison

(D)(Lunar) The vehicle will be capable of cargo jettison at any point during the mission (including all phases of the lunar descent).

3.02.03.01.05 Space-Based Crew Support Duration (LOR)

(D)(SB & LOR) The space-based vehicle will be capable of supporting the crew for up to 26 days in case of mission abort.

3.02.03.01.06 Space-Based Crew Support Duration (LOD)

(D)(SB & LOD) The space-based vehicle will be capable of supporting the crew for up to 18 days in case of mission abort.

3.02.03.02 Failure Tolerance

3.02.03.02.01 Crew Safety

(C)(Piloted) Critical functions affecting crew safety will be two failure tolerant.

3.02.03.02.02 Mission Support

(C) Critical mission support functions will be one failure tolerant.

3.02.03.02.03 Noncritical Functions

(C) Noncritical functions will be zero failure tolerant.

3.02.03.03 Quality Assurance

3.02.03.04 Software Product Assurance

3.02.03.05 Maintainability

3.02.03.05.01 LEO Transportation Node

(C)(SB) The vehicle will be maintained, mated, and serviced at Space Station Freedom.

3.02.03.05.02 Line Replaceable Unit

(C) The vehicle design will be capable of removal and replacement of units at the functional component level.

3.02.03.05.03 Checkout Tests

(C) The vehicle will provide for checkout tests of critical functions.

3.02.03.05.04 Unit Accessability

(C) The vehicle will incorporate units to be maintained through telerobotic or EVA servicing external to the pressurized environment.

3.02.04 Environmental Conditions

3.02.04.01 Natural Environments

3.02.04.01.01 Unprepared Landing Surface

(C)(Lunar) The unpiloted cargo and first piloted mission vehicles will be capable of landing on a surface with a 15-degree slope and 1.0m irregularity.

3.02.04.01.02 Prepared Landing Surface

(C)(Lunar) The vehicle will be capable of landing on a surface with 2-degree slope, 0.2m irregularity, and 50m diameter.

3.02.04.02 Induced Environments

3.02.05 Transportability

3.03 Design and construction

3.04 Logistics

3.05 Personnel and Training

3.06 Interface Requirements

3.06.01 Launch Vehicle

3.06.01.01 Shroud Diameter

3.06.01.01.01 Space-Based Shroud Diameter

(P)(SB) The vehicle will be compatible with a launch vehicle shroud 10 meters in diameter and 30 meters in length.

3.06.01.01.02 Ground-Based Shroud Diameter

(P)(GB) The vehicle will be compatible with a launch vehicle shroud 10 meters in diameter and 30 meters in length.

3.06.01.02 Launch Site

(C) The vehicle (or vehicle components for on-orbit assembly) will be launched from the Kennedy Space Center.

3.06.02 Planet Surface System Interfaces

(Lunar)

3.06.02.01 PSS Payload Support Services

3.06.02.01.01 Power

(C)(Lunar) The vehicle will supply the PSS payload with 2 kWe (average) and 3 kWe (peak) power during transit through 48 hours after lunar touchdown.

3.06.02.01.02 Heat Rejection

(C)(Lunar) The vehicle will supply the PSS payload with 1-kWt thermal rejection during transit through 48 hours after lunar touchdown.

3.06.02.01.03 Data Communications

(C)(Lunar) The vehicle will be capable of receiving a 200-kbps health and status monitoring data stream from the PSS payload during transit through 48 hours after lunar touchdown and transmitting that data to the mission operations center.

3.06.02.01.04 Payload Release Latches

(C) The vehicle will provide remote payload release latches to assist in the deployment of the PSS payloads.

3.06.02.02 Self-Support Duration

(D) (Piloted) The vehicle will be capable of self-support for up to 30 days on the lunar surface.

3.06.02.03 Planet Surface System Support to Piloted Vehicles

(Piloted)

3.06.02.03.01 Power

(C)(Piloted) The vehicle will consume $\leq$ 2 kWe of power (average) after lunar landing on missions greater than 30 days in duration.

3.06.02.03.02 Heat Rejection

(C)(Piloted) The vehicle will operate within boiloff limits with $\leq$3-kWt heat rejection and a thermal tent on missions greater than 30 days in duration.

3.06.02.03.03 Surface Maintenance Support

(D)(Piloted) The vehicle will allow access for maintenance operations on the lunar surface.

3.06.02.03.04 Data Communications

(C)(Piloted) The vehicle will provide a 200-kbps telemetry/command datalink to the PSS for health and status monitoring.

3.06.02.03.05 Navigation Aids

(C)(Piloted) Beginning with the first piloted mission, the vehicle will use navigation aids to assist in lunar landing.

3.07 Characteristics of Subordinate Elements

3.08 Precedence

3.9 Qualification

3.10 Standard Sample

3.11 Preproduction Sample

4.0 VERIFICATION

4.01 General

4.02 Quality conformance for inspection

5.0 PREPARATION FOR DELIVERY

6.0 NOTES

6.01 Intended use

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16. Abstract

This report presents the results of systems analyses and conceptual design of space transfer vehicles (STV). The missions examined included piloted and unpiloted lunar outpost support and spacecraft servicing, and unpiloted payload delivery to various earth and solar orbits. The study goal was to examine the mission requirements and provide a decision data base for future programmatic development plans. The final lunar transfer vehicles provided a wide range of capabilities and interface requirements while maintaining a constant payload mission model. Launch vehicle and space station sensitivity was examined, with the final vehicles as point designs covering the range of possible options. Development programs were defined and technology readiness levels for different options were determined. Volume I presents the executive summary, Volume II provides the study results, and Volume III the cost and WBS data.

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